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PROPULSION REQUIREMENTS FOR SOFT LANDING

In Extraterrestrial Landing Environments

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Propulsion Requirements for Soft Landing in Extraterrestrial Environments

Summary and Propulsion Design Guide

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FOREWORD

This document was prepared in compliance with the requirement for the final report for National Aeronautics and Space Administration contract NAS 7-124, "Propulsion Requirements for Soft Landing in Extraterrestrial Environments."

ABSTRACT

Volume I, "Propulsion Requirements for Soft Landing in Extraterrestrial Environments - Summary and Design Guide," presents the major results of studies conducted under NASA Contract NAS 7-124. Landing trajectory concepts applicable to landings on the moon, Mars, Venus, Mercury and the Earth are described. For the most suitable landing techniques, the required propulsive maneuvers are defined, and the optimum characteristics of propulsion systems for performance of these maneuvers are presented. Investigations to determine appropriate interplanetary trajectories upon which to base landing analyses and to evaluate takeoff propulsion requirements are discussed.

In the Design Guide, a brief summary of the characteristics of propulsion systems for performance of various maneuvers associated with the landing phase of extraterrestrial missions is presented. Certain qualitative aspects of the required propulsion systems are described in addition to a tabular presentation of optimum operating parameters. In addition, data illustrating the effect of variation of propulsion paramcters on vehicle velocity requirements and payload capabilities are included.

TABLE OF CONTENTS - VOLUME I

INTRODUCTION

The study, "Propulsion Requirements for Soft Landing in Extraterrestrial Environments" was conducted (1) to define the most suitable landing concepts for landings on Mars, Venus, Mercury, Earth and the moon, in order to specify the required propulsive phases, and (2) to determine the optimum characteristics of propulsion systems for these propulsive phases.

This volume presents a summary of the analyses conducted in investigating the various landing concepts and the propulsion system characteristics which provide optimum performance for the required propulsive maneuvers. The latter results are presented as a Propulsion Design Guide, and include, in addition to optimum system characteristics, sufficient parametric data to demonstrate the effect of nonoptimum operation on landing vehicle capabilities.

Analysis of lunar and planetary landings entailed initially the selection of appropriate transfer trajectories and consequent planetary arrival conditions; these results provided the applicable initial conditions upon which to base subsequent studies of landing maneuvers. The sequence of maneuvers comprising an extraterrestrial landing operation was dependent primarily on the presence or absence of an atmosphere about the destination body. As a result, the landing maneuver profiles were qualitatively, though not quantitatively, similar for the all-propulsive lunar and Mercury landings, and for the Earth, Mars and Venus landings, which utilized the atmospheres of those bodies for a major part of the required vehicle deceleration.

For a landing mission as defined in this study, the first in the chronological sequence of propulsive and aerodynamic maneuvers considered for terrestrial and extraterrestrial landing phase analyses was the propulsive terminal correction utilized to establish the initial conditions required for safe entry into a planetary atmosphere or deceleration into a prescribed planetocentric circular orbit. This maneuver, in preference to earlier (e.g., midcourse correction) or later (e.g., deceleration into orbit) maneuvers was chosen, first, because it is essential to satisfactory performance of any subsequent maneuvers, and second, because it is the earliest maneuver primarily influenced by the gravitational field of the destination planet.

The basic results of the study were the definition of the propulsive maneuvers associated with landings on each of the destination bodies, and specification of the velocity requirements and optimum propulsion system parameters for these maneuvers. Investigations in the area of aerodynamic landing vehicle trajectories were performed by General Dynamics/Astronautics.

These studies furnished aerodynamic entry corridor requirements for use in the terminal correction analyses and the general characteristics of atmospheric graze maneuvers for investigations of propulsive/aerodynamic orbit-establishment maneuvers.

SUMMARY

The study, "Propulsion Requirements for Soft Landing in Extraterrestrial Environments", was conducted (1) to define the most suitable landing concepts for landings on Mars, Venus, Mercury, Earth and the moon, in order to specify the required propulsive phases, and (2) to determine the optimum characteristics of propulsion systems for these propulsive phases.

Analysis of lunar and planetary landings entailed initially the selection of appropriate transfer trajectories and consequent planetary arrival conditions; these results provided the applicable initial conditions upon which to base subsequent studies of landing maneuvers. The sequence of maneuvers comprising an extraterrestrial landing operation was dependent primarily on the presence or absence of an atmosphere about the destination body. As a result, the landing maneuver profiles were qualitatively, though not quantitatively, similar for the all-propulsive lunar and Mercury landings, and for the Earth, Mars and Venus landings, which utilized the atmospheres of those bodies for a major part of the required vehicle deceleration.

For a landing mission as defined in this study, the first in the chronological sequence of propulsive and aerodynamic maneuvers considered for terrestrial and extraterrestrial landing phase analyses was the propulsive terminal correction utilized to establish the initial conditions required for safe entry into a planetary atmosphere or deceleration into a prescribed planetocentric circular orbit. This maneuver, in preference to earlier (e.g., midcourse correction) or later (e.g., deceleration into orbit) maneuvers was chosen, first, because it is essential to satisfactory performance of any subsequent maneuvers, and second, because it is the earliest maneuver primarily influenced by the gravitational field of the destination planet.

The basic results of the study were the definition of the propulsive maneuvers associated with landings on each of the destination bodies, and specification of the velocity requirements and optimum propulsion system parameters for these maneuvers. These data provide an indication of the system requirements for performance of various phases of landing missions, and in combination with takeoff data generated in this study and Earth-launch and midphase requirement data obtained previously, furnish sufficient information to permit analysis of overall vehicle requirements for interplanetary missions.

The need for a propulsive terminal correction maneuver (for all except lunar missions) is the result of the inability of launch and midphase systems to place the spacecraft in a trajectory sufficiently accurate to assure rendezvous with the desired region in space, i.e., within the boundaries

of the allowable entry corridor for subsequent aerodynamic landing or at a suitable location from which to initiate a propulsive orbit-establishment maneuver. In the latter case, a close tolerance is unnecessary, and for a final orbit altitude of 300 ± 30 nautical miles at Mars and Venus, terminal corrections on the order of 600 ft/sec were required, based on trajectory and correction execution errors obtained from a previous study (see Reference 1). For aerodynamic entries, the stringent corridor limits at Venus in comparison to relatively wide limits at Mars (with Earth being close to, but slightly less severe than Venus), results in rather modest terminal correction requirements for Mars missions, 300 to 400 ft/sec, and approximately a 4000 ft/sec terminal correction for the Venus mission considered. Terminal correction requirements for the Earth-return missions analysed ranged from 800 ft/sec to 3000 ft/sec. In those instances (two cases for Earth, one for Venus) where the correction velocity requirement exceeded 1000 ft/sec, a two-impulse correction scheme was analysed, and it was determined that a partial correction, applied at 100,000 nautical mile range, and a second correction, applied at sufficiently close range to provide the required accuracy, reduced the total correction velocity requirements to values between 300 ft/sec and 800 ft/sec.

Orbit establishment maneuvers can be performed entirely by propulsive means (as would have to be the case at the moon and Mercury), or by an aerodynamic deceleration phase (atmosphere graze) supplemented by propulsive impulses. Analysis of the former technique indicated that for lunar and interplanetary missions of interest, orbit-establishment maneuvers (or orbit-departure maneuvers, since the propulsion requirements are quite similar for thrust-to-weight ratios in the region of greatest interest) represent a major propulsive operation, requiring ideal velocity increments (ΔV) ranging from slightly in excess of 3000 ft/sec at the moon to more than 25,000 ft/sec at Mercury. For Earth-return orbit-establishment missions, ΔV is approximately 10,000 ft/sec from a lunar transfer trajectory, about 14,000 ft/sec for moderately low energy Mars or Venus transfers, and on the order of 22,000 ft/sec for higher-energy interplanetary trips. Propulsion system optimum thrust-to-weight ratio (F/W) was found to be strongly dependent on the amount of inert weight in the system per unit thrust. For representative values of vehicle parameters, optimum local F/W was generally between 0.3 and 0.5; however, use of F/W values between 0.2 and 0.7 had little effect on payload. Employment of atmosphere graze maneuvers at the Earth, Mars and Venus provided an efficient means of achieving approximately a 15 to 30 percent vehicle velocity reduction prior to performing a propulsive orbit-establishment maneuver. Several techniques for combining an aerodynamic graze with subsequent propulsive operations to inject a space vehicle into 300 nautical mile planetary orbit were considered. For any of the methods, the propulsive ΔV requirement was approximately the difference between the atmosphere exit velocity and the final circular velocity. The most efficient method was to permit the vehicle to coast to apoapsis (assuming, of course, that the atmosphere exit velocity was less than the local escape velocity), decelerate or accelerate) to generate a 300 nautical mile periapsis, coast to periapsis and then decelerate to orbital velocity. In those instances where the atmosphere exit velocity is above or close to escape velocity, a deceleration impulse at atmospheric egress can be utilized to limit apoapsis altitude (and therefore, coast time) to reasonable values.

Direct landing systems comprised of a propulsion device followed in sequence by an ablative heat shield for aerodynamic braking were analysed to determine if hypersonic entry can be accomplished more efficiently by a combined system or by the aerodynamic device alone. The study was conducted parametrically to circumvent the problem of the uncertainty of ablative shield weight requirements. For the shield weight vs velocity characteristic representing the current best estimate, the analysis indicated that the lightest overall landing vehicle is obtained if approach velocity above 56,000 ft/sec is cancelled propulsively.

A major propulsive maneuver is required to decelerate an orbiting vehicle to the lunar or Mercury surface. Evaluation of various landing trajectory concepts indicated that a thrust orientation profile in which the thrust vector is continuously aligned along the velocity vector, but oppositely directed, represents close to an optimum descent trajectory, and was therefore employed in the major portion of the required landing analyses. The optimum F/W for lunar descent-from-orbit was approximately 0.6 (in terms of Earth weight), and the ideal ΔV requirement was slightly less than 6000 ft/sec. For descent from Mercury orbit, an optimum thrust-to-Earth weight ratio of approximately 0.9 was obtained, and the ΔV requirement was approximately 10,900 ft/sec.

A study of the altitude and range errors experienced in reaching a desired lunar or Mercury landing point was conducted to determine the magnitude of terminal position discrepancies which occur if execution of the orbital descent maneuver deviates from nominal with respect to thrust level, ignition time and alignment of the thrust and velocity vectors. For representative error values, i.e., 1 percent from nominal thrust and 0.5 degree from parallel vector alignment, position errors as shown below were obtained.

	Lunar Landing		Mercury Landing	
	Δ altitude, ft	Δ range, n mi	Δ altitude, ft	Δ range, n mi
1 percent thrust error	1360	1.75	1560	2.40
0.5 degree alignment error	4450	0.10	5700	0.12

Each second of deviation from the nominal ignition time for orbital descent maneuvers resulted in a range error slightly less than one nautical mile at the moon and somewhat above 1.5 nautical miles at Mercury.

The propulsion requirements for near-surface translation maneuvers utilized to reach a desired landing site are dependent on the distance traversed and on the technique employed for performance of the maneuver. A ballistic flight to the desired site is the most economical from the standpoint of propellant expenditure, but it is the least safe and, because of possible high altitudes reached, might offer inadequate opportunities for surveillance. Continuous propulsion methods, both single engine and multiple-engine, were evaluated; the former is simpler and more economical, and was therefore considered in greater detail. Results indicated that 45 degrees was the optimum orientation during the acceleration and deceleration phases. For a 3000 feet lunar translation maneuver, approximately 400 ft/sec of ideal ΔV capability was required.

Vertical descent from the lunar translation maneuver termination point (on the order of several hundred feet above the surface) required a relatively modest propulsive capability, approximately 100 ft/sec, but this value was dependent on the propulsion system having a throttling capability of at least 10:1. With insufficient throttling ability, penalties on the order of an additional 100 ft/sec can be incurred.

The use of a small retrorocket in conjunction with a parachute and impact device for the terminal phase of Earth, Mars, and Venus landings offers a means of circumventing the sizable weight penalty associated with designing parachutes for very low (approximately 20 ft/sec) terminal velocities. For low (less than 25 ft/sec) impact velocities, the minimum weight system includes each of the three component parts and has the characteristics shown below.

	Parachute Terminal Velocity, ft/sec	Rocket F/W	Impact Velocity, ft/sec	Percent of Landed Weight
Earth	70	1.5	10	4.4
Earth	65	1.2	25	4.1
Mars	120	0.93	10	5.0
Mars	120	0.85	25	4.8
Venus	43	1.20	10	3.2
Venus	40	0.80	25	2.7

If impact velocity is unrestricted by stability or landing gear design problems, the optimum impact velocities are 55, 35 and 42 ft/sec at Earth, Mars and Venus respectively, and only at Mars does the rocket remain as a component of the minimum-weight system. The relatively high optimum values of impact velocity result from the excellent energy-absorption capabilities of various crushable or frangible materials utilized in landing gear legs.

Takeoff-to-orbit maneuvers at the moon, Mercury, Mars and Venus were evaluated to determine propulsion requirements for use in subsequent overall vehicle analyses. Ideal ΔV requirements were approximately 6000 ft/sec for the moon (50 n mi orbit), 11,200 ft/sec for Mercury, 15,000 ft/sec for Mars and 40,000 ft/sec for Venus (the latter three are for 300 n mi orbits). Optimum F/W values were on the order of two to three in terms of local weight values for the moon, Mercury and Mars, but substantially less for Venus. The drag effects caused by the dense atmosphere on that body nullify the benefits of high acceleration which are obtained elsewhere. The optimum takeoff thrust-to-Venus weight was approximately 1.3.

The analyses of various propulsive maneuvers yielded a broad spectrum of propulsion system requirements, ranging in velocity increment from below 5000 ft/sec to above 25,000 ft/sec, in optimum Earth F/W from below 0.1 to above 2 (though the latter occurs only for takeoff stages and is not actually of concern in the present study), and in propellant storage requirements from a few days to several years. These variations, and a vehicle size range covering two order of magnitude, were included in a broad parametric study of optimum propulsion system parameters. A brief summary of results, representing several selected cases, is presented below.

System Type	Operating Parameter	Allowable Parameter Increment Percent Payload Loss 0.5	1.0	Optimum Value
O_2/H_2 Pressure-Fed	P_c, psia	+25	+35	55
	ϵ	-80	-100	200
	MR	+0.9	±1.2	6.20
F_2/H_2 Pressure-Fed	P_c	±22	±35	80
	ϵ	- 60	-90	200
	MR	±2.3	-3.8	16.90
NTO/50-50 Pressure Fed	P_c	+40	+60	130
	ϵ	-110	-145	310
	MR	±0.13	±0.20	2.18
O_2/H_2 Pump-Fed	P_c	-450	-550	1360
	ϵ	-200	-275	450
	MR	±1.0	±1.4	6.90
F_2/H_2 Pump-Fed	P_c	-650	-900	1770
	ϵ	-230	-270	410
	MR	-3.2	-4.6	17.35
NTO/50-50 Pump-Fed	P_c	-600	-800	1800
	ϵ	-175	-230	371
	MR	±0.12	±0.18	2.21

Mission: ΔV = 14,000 ft/sec, F/W = 0.3

Of particular importance is the wide range over which optimum parameters can be varied without seriously penalizing payload capabilities.

A supplementary study of three selected interplanetary missions was performed to translate the propulsion requirements determined in this and other related studies into overall vehicle requirements. The first mission, with the relatively modest objective of soft landing a 2000 pound unmanned probe on Mercury, at an optimum transfer opportunity that occurs in 1973, and again in 1986, required a launch weight of 12,910,000 pounds. In this instance, there are no maneuvers which can be performed aerodynamically. Two far more ambitious missions, manned excursions to Mars and Venus (50,000 pound payload in each case), yielded launch vehicle weights of 118,200,000 pounds and 2,496,000,000 pounds respectively. These latter values indicate the greater facility for performance of a manned Mars landing and return mission in comparison to a manned Venus landing and return mission. If the assumptions of a direct mission profile (in contrast to use of an excursion vehicle), current chemical propulsion and a 50,000 pound payload requirement are reasonable, then the results demonstrate that the Earth launch weight required for a manned Mars landing and return mission is an order of magnitude greater than is required for the Apollo lunar mission.

LANDING MISSION CONCEPTS

FACTORS AFFECTING LANDING ANALYSIS

A lunar or interplanetary round-trip mission is comprised of a sequence of closely interrelated propulsive and nonpropulsive phases which can be described broadly by the chart in Figure 1 . The objective of this program was an investigation of terrestrial and extraterrestrial landings; however, the landing investigations require analysis or review of the mission phases which precede and follow the landing phase to provide adequate data for comprehensive investigation of the landing phase. Therefore, with the discussions of the discrete phases of planetary landings, the necessary descriptions of interplanetary trajectories and planetary takeoff requirements are included.

For each planet, the landing mission is characterized by a sequence of maneuvers; the nature of these maneuvers is governed primarily by the presence or absence of an atmosphere about the subject planet. For example, a lunar landing must be entirely propulsive and therefore entails a major deceleration phase, either from orbit or from a transfer trajectory, a hover/translation phase and a vertical descent phase, all of which are rocket-powered. The corresponding portions of a Mars landing utilize aerodynamic vehicle drag for most of the required velocity cancellation, parachute drag for further deceleration and maneuvering, and possibly a small rocket for a final small amount of deceleration prior to impact.

LANDING MANEUVERS

The major deceleration phase of an extraterrestrial landing may be accomplished in a single maneuver directly from the approach trajectory or by a sequence of two maneuvers in which the vehicle first decelerates into orbit about the destination body and subsequently descends to the surface. The approach velocity can range from a value slightly in excess of the local parabolic velocity as for an Earth-return trajectory from the moon, to several times parabolic velocity, as for fast, hyperbolic, interplanetary trajectories.

Three major types of direct landing are possible.

1. Direct Vertical Landing

   For this type of landing the vehicle approaches the destination planet along a vertical flight path. Propulsion is applied at the correct altitude to brake the vehicle for the landing or, if the destination planet has an atmosphere, aerodynamic drag can be used to decelerate the vehicle.

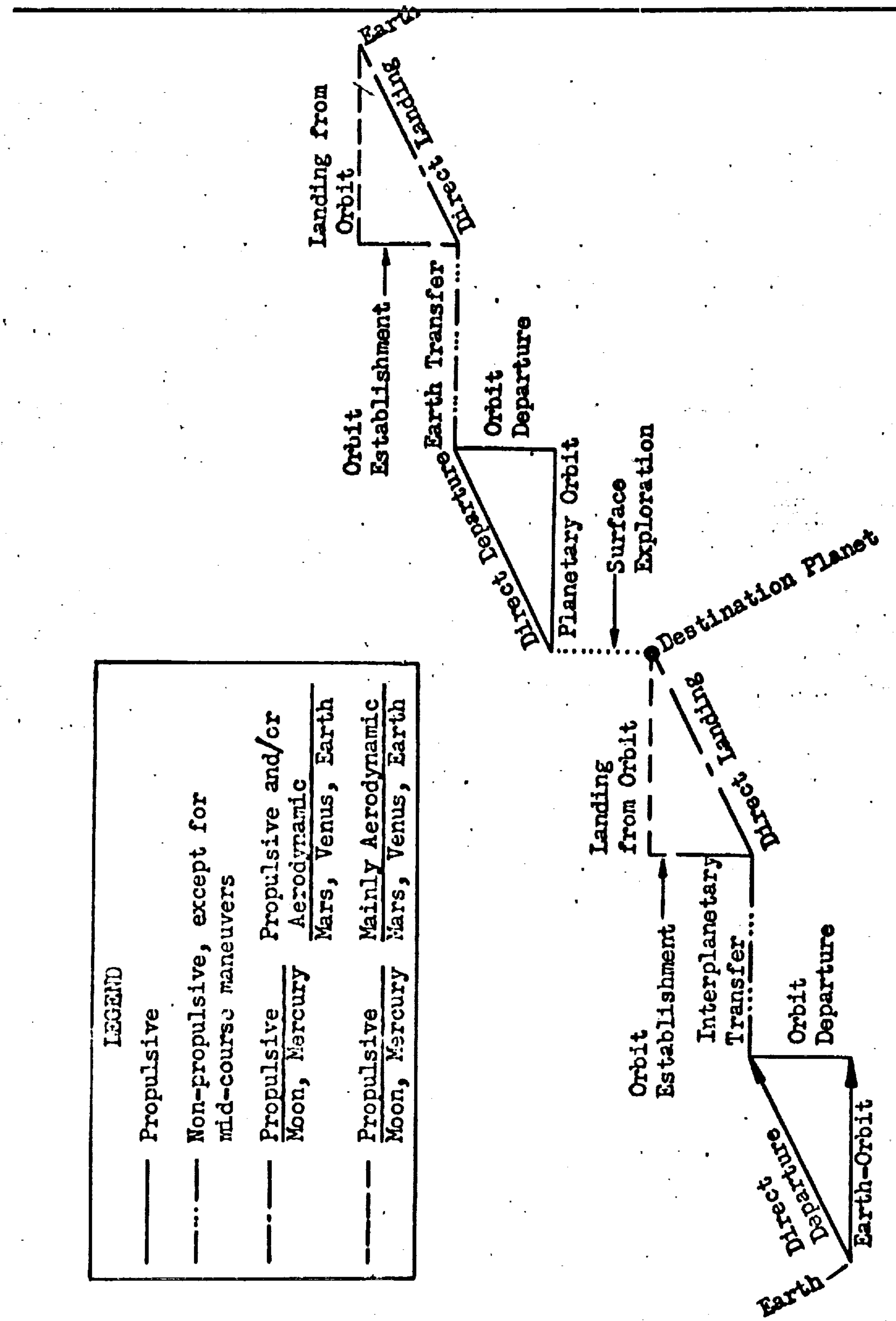


Figure 1 . Interplanetary Missions

2. Direct Nonvertical Landing

The vehicle approaches on a parabolic or hyperbolic path which is somewhat displaced from a vertical landing approach trajectory, and would, in the absence of a planetary atmosphere or a propulsive braking phase, bypass or impact obliquely, the destination planet. Propulsion or aerodynamic braking slows the vehicle for the landing. The nature of the approach flight path is selected to correspond to the characteristics of the landing vehicle.

3. Grazing Approach Landing

This landing trajectory type is for aerodynamic braking only. The approaching vehicle grazes the planetary atmosphere, and then, slowed by drag during the graze, again leaves the atmosphere. Subsequently, the vehicle may circle the planet in an elliptical orbit before again entering the atmosphere or, if it has been decelerated sufficiently, re-enter after only a short skip out of the atmosphere. One or more grazes may be necessary before the vehicle is slowed to a velocity suitable for the final braking entry.

The direct vertical method offers guidance simplicity as its principal advantage, but abort capability is poor, gravity losses are high for a propulsive braking maneuver, and heating rates and deceleration levels are high for an aerodynamic braking maneuver; the alternative techniques are therefore preferred.

The two principal types of orbit-establishment maneuver are:

1. Direct Orbit Establishment

In this maneuver, the vehicle, when it is in the vicinity of the target planet, is propulsively slowed to orbit velocity.

2. Grazing Approach Establishment

In this maneuver, the vehicle grazes the atmosphere of the target planet. After the graze, the vehicle, slowed by drag during the graze, leaves the atmosphere. One or more of these grazes can be used to decelerate the vehicle so that it leaves the atmosphere with about the velocity of a low altitude planetary orbit. After the final graze, a short propulsive phase is utilized to establish a suitable orbit above the atmosphere.

The two major descent-from-orbit methods are:

1. Orbit Decay Landing

   If the orbit altitude is sufficiently low, the vehicle experiences aerodynamic drag, and the altitude of the orbit is slowly decreased. Subsequently, the vehicle enters sufficiently dense atmosphere to introduce a period of high deceleration, and the vehicle is slowed for landing.

2. Direct Landing

   In this landing concept, the vehicle is braked in orbit propulsively to initiate descent. If the planet has an atmosphere, the vehicle can enter the atmosphere and perform an aerodynamic landing. If no atmosphere is present, a propulsion phase must be used to slow the vehicle for landing.

It is more difficult to land at a particular site by the orbit decay method. Though this technique provides lower heating rates and deceleration levels, the values of these parameters are not particularly high for direct orbit-descent. The direct method is therefore preferred.

DECELERATION METHODS

Deceleration techniques were reviewed to establish the factors pertinent to the analysis of various landing methods. Aerodynamic, propulsive and combined propulsive/aerodynamic braking systems were considered. Analysis of the landing trajectory for an aerodynamic landing vehicle requires determining a suitable entry corridor in which vehicle deceleration is sufficient to prevent skipping out of the atmosphere but not beyond the tolerance limits of manned or unmanned payloads. The entry corridor can be defined either by the use of entry angle (angle of the vehicle velocity vector) at a specified altitude, or by virtual periapsis (the periapsis that the entry conic would have if there were no planetary atmosphere). The undershoot boundary (lowest periapsis or highest entry angle) and the overshoot boundary (highest periapsis or lowest entry angle) are the boundaries of the entry corridor. The entry corridor can then be described by an entry angle range or by a corridor depth (the difference between the virtual periapsis of the overshoot and undershoot boundaries).

A comparison of the entry problems at Mars, Venus, and the Earth is presented in Table 1 . In Table 1 are compared on a relative basis, total heating (q), maximum heating rate ($\dot{q}$), maximum deceleration (G), and entry corridor width (h) for the three planets of interest in the present study.

TABLE 1

	Satellite Entry			Parabolic Entry			
	G	$\dot{q}$	q	G	h	$\dot{q}$	q
Earth	1	1	1	1	1	1	1
Mars	0.4	0.1	0.2	1	12	0.5	0.2
Venus	0.9	0.8	0.9	1	1	0.9	1

The similarity of the aerodynamic entry problems for Earth and Venus is indicated in Table 1 . Entry at Mars, however, is less difficult due to the atmosphere of Mars (which has a lower density variation with altitude than the atmospheres of Venus and Earth) and the lower gravity of Mars.

Analysis of propulsive landing maneuvers entails primarily the determination of ideal velocity requirements for the type of trajectory selected; this quantity is dependent upon the vehicle thrust-to-weight ratio, the vehicle thrust magnitude and orientation program, and the type of propulsion system being considered. Various propulsion system studies have indicated that, (1) a tangential thrust program is an efficient method of velocity reduction and is a logical choice for most propulsive braking maneuvers, (2) engine operation at maximum thrust (no throttling) minimizes gravity losses during the propulsive maneuver and consequently would be used in most propulsive braking maneuvers, and (3) restarts should be avoided whenever possible to increase system reliability.

During entry into a planetary atmosphere at supersatellite velocity, vehicles experience high heating rates and decelerations. In order to reduce heating, and thereby, heat shield requirements, it may be necessary (or desirable from the payload standpoint) for a propulsive phase to precede the aerodynamic entry. For most efficient use of a propulsion system, a propulsive phase should occur when the vehicle has its highest velocity, or just before aerodynamic braking begins. The propulsive phase of a combined propulsive/aerodynamic landing therefore takes place just above the planetary atmosphere. For combined propulsive/aerodynamic braking, an optimization must be conducted to determine the distribution of the total vehicle velocity reduction between the propulsive and aerodynamic phases. Except for this optimization, each phase should not appreciably influence the other. The most likely areas of propulsion system application are summarized in Tables 2 and 3 .

TABLE 2

LANDING ON PLANETS WITH NO ATMOSPHERE

	Trajectory Concept	Areas Requiring Propulsion
1.	Direct Nonvertical Landing from Supersatellite Velocity	Approach Trajectory Correction Major Braking Near-surface Maneuvering
2.	Direct Orbit Establishment from Supersatellite Velocity	Approach Trajectory Correction Major Braking Orbit Correction
3.	Direct Landing from Satellite Velocity	Deorbiting Major Braking Near-surface Maneuvering

TABLE 3

LANDING ON PLANETS WITH ATMOSPHERE

	Trajectory Concept	Vehicle Type	Areas Requiring Propulsion
1.	Direct Nonvertical Landing from Supersatellite Velocity	Lifting Body of Ballistic	Approach Trajectory Correction Braking (Prior to Aerodynamic Entry to Reduce Heating and/or Deceleration)
2.	Grazing Approach Orbit Establishment	Lifting Body	Approach Trajectory Correction Braking (Prior to Aerodynamic Entry to Reduce Heating and/or Deceleration) Orbit Establishment (After Graze) Orbit Correction
3.	Direct Landing from Satellite Velocity	Ballistic Airplane	Deorbiting Deorbiting Propulsion for Conventional Aircraft Flight Below Orbital Velocity

EARTH RETURN MISSIONS

ATMOSPHERIC ENTRY AND TERMINAL CORRECTION REQUIREMENTS

The trajectory of a space vehicle approaching a planet defines the conditions at atmospheric entry. Three round trip missions, two to Mars and one to Venus, were selected for investigation of terminal corrections. These missions, representative of relatively short transfer time trajectories, encompass the range of Earth entry conditions that can presently be anticipated for interplanetary missions of the future. Trajectory details for the missions are presented in Table 4 .

TABLE 4
EARTH ATMOSPHERIC RE-ENTRY MISSIONS

Trajectory	Launch Date	Trip Time, days	Hyperbolic Arrival Velocity, ft/sec	Asymptotic Approach Distance, n mi
Mars-Earth (2)	10 Nov 1971	278	43,500	4920
Mars-Earth (4)	26 Aug 1971	110	29,000	8360
Venus-Earth (6)	31 Dec 1965	86	12,650	12,920

Atmospheric Entry

The major parameters for atmospheric entry are entry velocity, incident angle at which a vehicle enters the atmosphere and the vehicle design. (For analysis purposes, a specific altitude above the effective atmosphere was defined to provide a basis for specification of entry corridor parameters; for Earth this altitude is 400,000 feet.) If the entry angle is too high for the entry velocity, an "undershoot" occurs where the atmospheric entry results in a higher deceleration rate than allowable. At the other extreme, too shallow an entry angle results in an "overshoot" where the atmospheric deceleration is insufficient, thus allowing the space vehicle to skip out of the atmosphere.

Analyses conducted by General Dynamics/Astronautics, using simulated reentry trajectories, were made to define those entry trajectories that are within specified maximum deceleration g limits. Aerodynamic landing configurations of the ballistic (Mercury Capsule), lifting body and airplane(Dynasoar) types were considered.

Examination of entry trajectory analysis results indicated that the ballistic entry vehicle design has the most stringent entry corridor requirements. Since this is a realistic system design, and entry vehicle design analysis has not progressed to the point of selection of optimum design concepts, the entry requirements for this vehicle were selected to determine terminal correction requirements. The Earth entry corridor for a drag vehicle is described in Figure 2 .

Present state-of-the-art accuracies of tracking, guidance, and propulsion prevent establishment, at planetary departure, of a space transfer trajectory sufficiently accurate to achieve the desired arrival conditions. Therefore, midcourse corrective propulsion maneuvers are a requisite. However, the midcourse correction is itself subject to inaccuracies in tracking and location and in maneuver execution. Midcourse corrections for the selected space missions were reviewed based on methods developed at Rocketdyne for NASA contract NAS 7-88, "Space Transfer Propulsion," described in Reference 1 .

In each mission, because of the various errors in the final midcourse corrective maneuver, the actual planetary approach hyperbola was not the desired one; deviations in the desired asymptotic approach distances existed at completion of the midcourse correction program (Table 5). The deviation translated to an atmospheric entry condition outside the defined entry corridor.

TABLE 5

PLANETARY ARRIVAL CONDITIONS

Trajectory	Hyperbolic Arrival Velocity (V_∞), ft/sec	Nominal Asymptotic Approach Distance (D), n mi	Deviation in Asymptotic Approach Distance (D), n mi	Actual Asymptotic Approach Distance (D_a), n mi
2	43,500	4580	340	4920
4	29,000	5660	2700	8360
6	12,650	10,500	2420	12,920

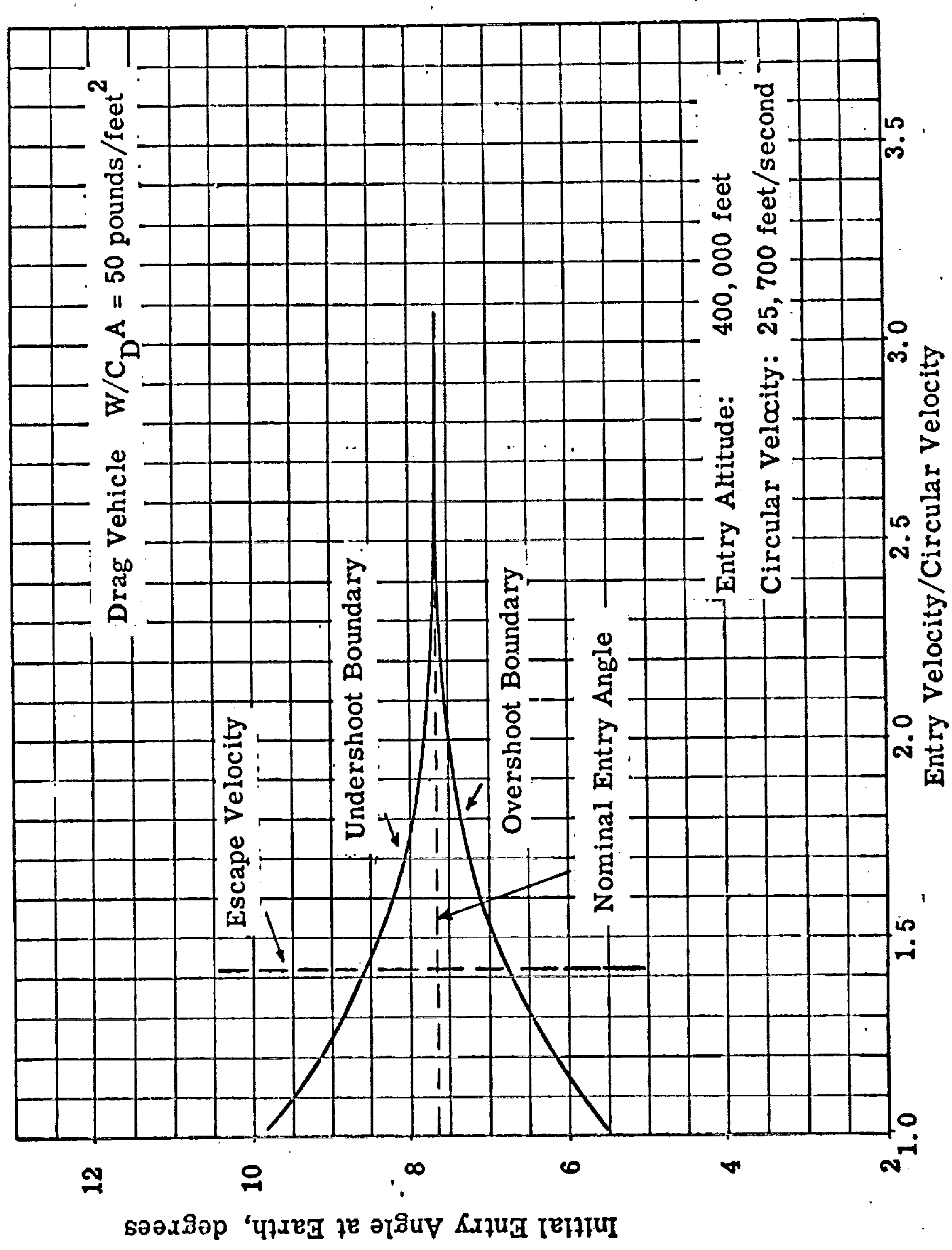


Fig. 2 Earth Entry Corridor

Terminal Correction Maneuvers

Because of these deviations, either additional corrective maneuvers, (terminal maneuvers) applied in the proximity of the target planet to ensure that entry corridor constraints are met, or a major propulsive deceleration phase (prior to atmospheric entry), which would enlarge the entry angle limits, was found to be required. For the latter technique, the propulsive deceleration required to increase the angle limits appreciably can be on the order of several thousand ft/sec; the investigation was, therefore, directed at determining the velocity required for a terminal correction maneuver which would result in a satisfactory entry. The optimum range at which to apply the terminal corrective maneuver and the deviation from nominal entry conditions resulting from errors encountered in executing terminal corrections were also evaluated.

In making the terminal correction, the velocity changes were assumed to be impulsive; the impulsive velocity assumption is based on Reference 1 analysis which indicates it is valid for the correction distances from the planet and velocity magnitudes involved.

Based on the asymptotic approach distance achieved as a result of the final midcourse correction, maneuver requirements for Mission (2) using a single terminal correction are presented in Figure 3 as a function of the range at correction and the deviations in asymptotic approach distance from the nominal.

Errors in terminal corrections (position and velocity-measurement errors, tracking and propulsive-maneuver execution errors) must be considered. The errors in measurement are range-dependent whereas errors in correction mechanization are a function of the magnitude of the correction velocity increment and the thrust-to-weight ratio of the system.

Considering use of a system with F/W ratios in the region between 0.1 to 0.5, the deviations in the vehicle velocity and angle at atmospheric entry are presented along with the correction velocity increment in Figure 4 as a function of range. The terminal correction errors have very little effect on changing the entry velocity; the parameter significantly affected by terminal correction errors is the entry angle. The range at correction and the ΔV to achieve entry within the corridor limits are tabulated in Table 6 for the three trajectories.

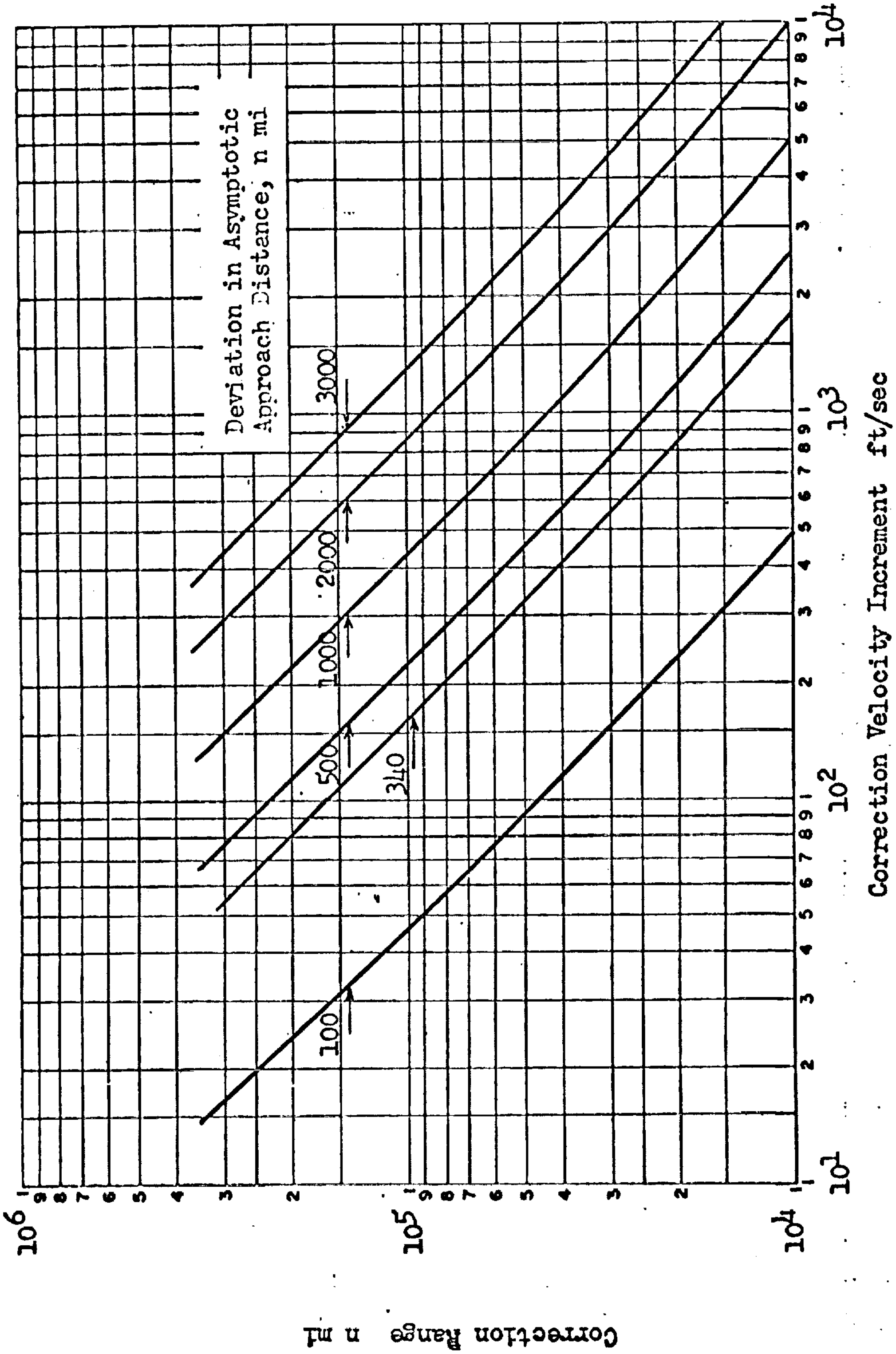


Fig. 3 The Effect of a Deviation in Asymptotic Approach Distance on the Terminal Correction Velocity Increment for Earth Atmospheric Re-entry. (Trajectory 2)

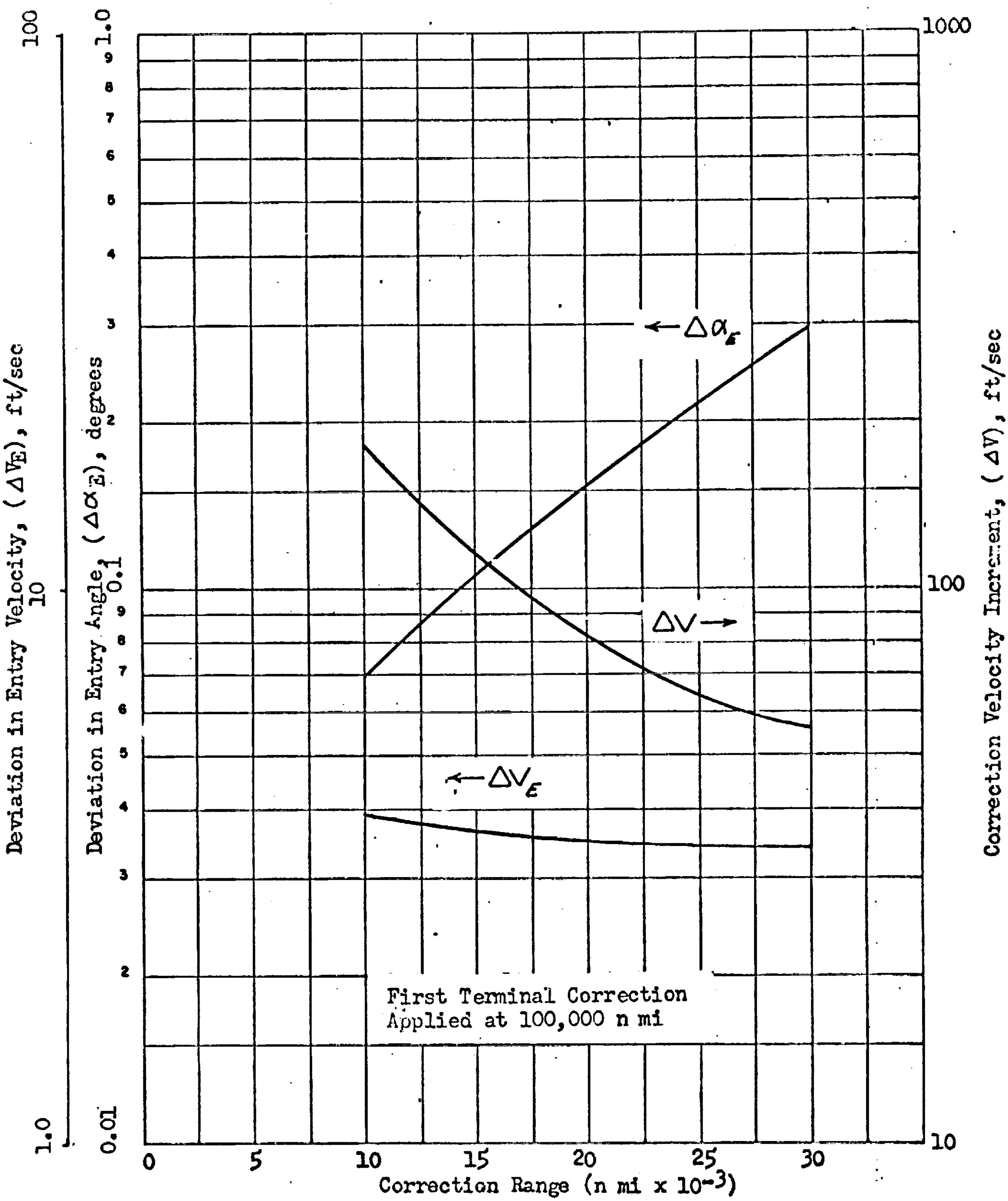


Fig. 4 Second Terminal Correction for Earth Atmospheric Re-Entry. (Trajectory 2)

TABLE 6

SINGLE TERMINAL CORRECTION FOR

EARTH ATMOSPHERIC REENTRY

Trajectory	Entry Corridor Half-band Width, degrees	Correction Range, n mi	Correction Velocity Increment, ft/sec
2	0.1	14,500	1,200
4	0.3	26,000	3,000
6	0.7	40,500	810

The velocity increments of trajectories 2 and 4 were considered to be excessive. Thus, the use of two terminal correction maneuvers was investigated for these trajectories. The first correction was made at the range of 100,000 nautical miles to reduce the velocity increment and yet stay within the realm of terminal corrections. This correction does not achieve the desired entry corridor limits. The range and velocity requirement for a second correction, based on satisfying the entry corridor limits, were determined. The velocity requirements are:

Trajectory			Single Correction Method Velocity Increment, ft/sec
2			1200
4			3000
			Dual Correction Method
	1st Correction	2nd Correction	Total Correction, ft/sec
2	160	125	285
4	760	60	820

A similar reduction in velocity requirement could be obtained by the use of a 2-correction scheme for trajectory (6); in that case, however, the velocity requirement for a single correction is reasonably small (810 ft/sec), and the possible propellant saving probably does not warrant the addition of need for engine restart capability imposed by utilization of a 2-correction technique.

Terminal Correction Results

The analysis demonstrated the need for terminal corrections if the selected entry corridor requirements are to be satisfied. Although use of dual terminal corrections involves restarting an engine, the sizable reduction in correction velocity increment obtained justifies employment of the technique.

The terminal correction analysis results for atmospheric entry of drag vehicles are summarized in Table 7. The range for applying a single correction or the second correction of dual corrections was specifically selected to restrict the deviations about the nominal entry angle to values equalling entry corridor half-band widths. The results are valid for F/W in the range of 0.1 to 0.5.

Based on these results, the use of terminal-correction maneuvers (single or dual as required) will provide the entry corridor required without a major deceleration propulsion phase (which would increase corridor width) prior to atmospheric entry.

TABLE 7

SUMMARY OF TERMINAL CORRECTIONS FOR EARTH ATMOSPHERIC REENTRY

Trajectory	Entry Corridor Half-band Width, degrees	Number of Terminal Corrections	Deviation in Entry Angle, degrees	Range at Correction, n mi	Total Terminal Correction ΔV, ft/sec
2	0.1	2	$\pm$ 0.1	100,000 14,000	285
4	0.3	2	$\pm$ 0.3	100,000 28,000	820
6	0.7	1	$\pm$ 0.7	40,500	810

PROPULSIVE EARTH ORBIT ESTABLISHMENT AND DEPARTURE MANEUVERS

The establishment of planetocentric orbits following an interplanetary transit represents a principal objective in early exploratory missions and an important intermediate step in many later extraterrestrial landing missions. The propulsion requirements for a range of mission hyperbolic excess velocities, and the effects of thrust-to-weight ratio and specific impulse were computed in a previous study; from the resulting data the optimum thrust-to-weight ratios and resultant payload-to-weight ratios for orbit establishment (and departure) maneuvers have been determined. The optimization was based on a tradeoff between the increased engine weight but decreased gravitational losses of high thrust-to-weight systems, and the decreased engine weight but increased gravitational losses of low thrust-to-weight systems. The effects of specific impulse (I_s), hyperbolic excess velocity (V_h), thrust-dependent weight factor (K_E) and propellant-dependent weight factor (K_T) on thrust selection was also determined.

The parameters selected for analysis of Earth-orbit establishment and departure maneuvers were varied over a sufficiently wide range to include many types of propulsion systems. The parameter, (K_E), varied from a value representative of a pump-fed system to a high value indicative of a redundant pump- or pressure-fed system. The parameters, (K_T) and I_s, had values typical of Earth-storable and cryogenic propellants. The value of V_h ranged from zero (appropriate for a lunar mission) to velocities required for interplanetary mission.

The results, shown in Table 8 , indicate that K_E has a small effect on payload whereas K_T, I_s, and V_h all affect payload considerably. K_E has the most pronounced effect on optimum F/W, with increasing K_E resulting in decreased optimum F/W.

For Earth-departure missions, a nonredundant propulsion system is likely to be employed (abort is relatively simple in the early phases of the mission); thus for a representative K_E of 0.025, the optimum F/W is approximately 0.37 for noncryogenic propellant systems and 0.45 for cryogenic systems. The optimum F/W for a redundant, cryogenic system (a representative K_E of 0.05) for the orbit-establishment mission is approximately 0.35. In both cases F/W values between 0.2 and 0.5 can be utilized with payload penalties on the order of only 1 percent.

At 30,000 ft/sec hyperbolic excess velocity, the propulsion requirement for orbit-departure is sufficiently high to warrant the use of 2 stages or staged propellant tanks. Results of the study of 2-stage and tank-staged systems are summarized in Tables 9 and 10. Investigation of the thrust requirements for a two-stage system indicated that F/W of 0.5 would be a near-optimum choice for both stages, but a large deviation can be permitted without introducing a significant payload penalty.

TABLE 8

SUMMARY OF THRUST OPTIMIZATION RESULTS FOR EARTH ORBIT-ESTABLISHMENT AND DEPARTURE MANEUVERS

			Thrust-Dependent Weight Factor (K_E) = 0.025				Thrust-Dependent Weight Factor (K_E) = 0.050			
Specific Impulse, Seconds	Hyperbolic Excess Velocity, ft/sec (V_h)	Propellant-Dependent Weight Factor (K_T)	Optimum Payload-to Gross Weight Ratio (PL/WG_{max})	Lower Thrust-to-Weight Ratio for 1-percent Payload Loss $(F/W)_L$	Optimum Thrust-to-Weight Ratio	Higher Thrust-to Weight Ratio for 1-percent Payload Loss $(F/W)_H$	Optimum Payload-to-Gross Weight Ratio (PL/WG_{max})	Lower Thrust-to-Weight Ratio for 1-percent Payload Loss $(F/W)_L$	Optimum Thrust-to-Weight Ratio	Higher Thrust-to Weight Ratio for 1-percent Payload Loss $(F/W)_H$
320	0	0.08	0.319	0.24	0.36	0.56	0.312	0.18	0.27	0.39
320	0	0.16	0.268	0.25	0.35	0.53	0.263	0.22	0.27	0.38
320	15,000	0.08	0.200	0.25	0.38	0.61	0.192	0.20	0.29	0.40
320	15,000	0.16	0.142	0.27	0.39	0.60	0.133	0.24	0.29	0.36
320	30,000	0.08	0.041	0.31	0.37	0.43	0.032	0.28	0.30	0.33
420	0	0.08	0.420	0.26	0.43	0.72	0.411	0.22	0.31	0.45
420	0	0.16	0.378	0.28	0.45	0.70	0.368	0.22	0.33	[illegible]
420	15,000	0.08	0.305	0.30	0.39	0.57	0.296	0.22	0.30	0.47
420	15,000	0.16	0.255	0.28	0.40	0.45	0.245	0.24	0.33	[illegible]
420	30,000	0.08	0.124	0.36	0.45	0.60	0.114	0.29	0.36	0.45
420	30,000	0.16	0.061	0.42	0.53	0.60	0.050	0.32	0.37	0.42

			Thrust-Dependent Weight Factor (K_E) = 0.075			
Specific Impulse, Seconds	Hyperbolic Excess Velocity, ft/sec (V_h)	Propellant-Dependent Weight Factor (K_T)	Optimum Payload-to Gross Weight Ratio (PL/WG_{max})	Lower Thrust-to-Weight Ratio for 1-percent Payload Loss $(F/W)_L$	Optimum Thrust-to-Weight Ratio	Higher Thrust-to Weight Ratio for 1-percent Payload Loss $(F/W)_H$
320	0	0.08	0.305	0.15	0.24	0.34
320	0	0.16	0.255	0.18	0.23	0.31
320	15,000	0.08	0.185	0.16	0.25	0.35
320	15,000	0.16	0.128	0.21	0.25	0.29
320	30,000	0.08	0.025	0.25	0.27	0.32
420	0	0.08	0.404	0.19	0.27	0.36
420	0	0.16	0.361	0.20	0.28	0.38
420	15,000	0.08	0.289	0.21	0.28	0.37
420	15,000	0.16	0.238	0.22	0.28	0.38
420	30,000	0.08	0.106	0.25	0.30	0.37
420	30,000	0.16	0.042	0.29	0.32	0.36

TABLE 9

ORBIT DEPARTURE VEHICLE OPTIMUM THRUST-TO-WEIGHT RATIOS

MISSION: 30,000 ft/sec HYPERBOLIC EXCESS VELOCITY ORBIT DEPARTURE

Vehicle	Optimum Initial Thrust-to-Weight Ratio	Initial Thrust-to-Weight Ratio Range for a Minus 2-Percent Payload
Single Stage	0.5	0.34 → 0.78
Two-Stage		
$F/W_2 = 0.5\ F/W_1$	0.65	0.64 → 1.22
$F/W_2 = F/W_1$		0.44 → 1.04
$F/W_2 = 2\ F/W_1$		0.36 → 0.64
∞ Tank Staging	0.46	0.28 → 0.79

TABLE 10

ORBIT DEPARTURE VEHICLE IDEAL VELOCITY INCREMENT AND PAYLOAD

MISSION: 30,000 ft/sec HYPERBOLIC EXCESS VELOCITY ORBIT DEPARTURE

Vehicle	Ideal Velocity Requirement, ft/sec	Relative Payload-to-Gross Weight Ratio	Percent of Single Stage Payload
Single Stage ($F/W = 0.5$)	21,700	0.110	100
Two-Stage ($F/W_1 = 0.65$) ($F/W_2 = 0.65$)	21,720	0.138	125
Single Stage Tanks Jettisoned One Time ($F/W = 0.5$)	21,680	0.134	122
Single Stage Tanks Jettisoned 4 Times ($F/W = 0.5$)	21.670	0.146	133

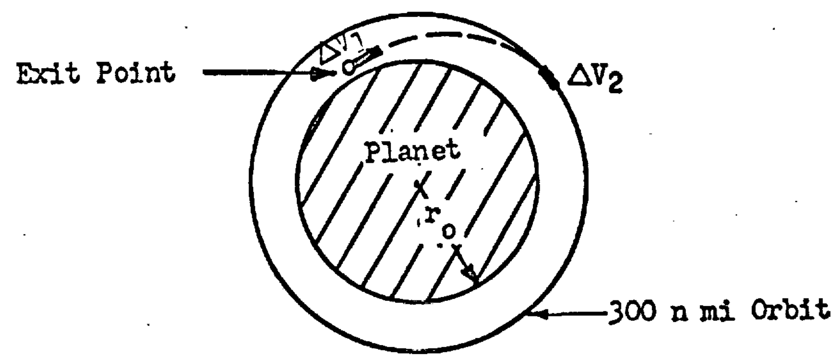
EARTH ATMOSPHERIC GRAZE MANEUVERS

To reduce the propulsive velocity requirements for establishing an Earth orbit from an interplanetary return trajectory, consideration has been given to first decelerating the vehicle by the aerodynamic drag provided by an atmospheric grazing maneuver (in which the vehicle "skims" the Earth's upper atmosphere). Following the graze, a propulsive maneuver is performed to establish a 300 n mi circular orbit.

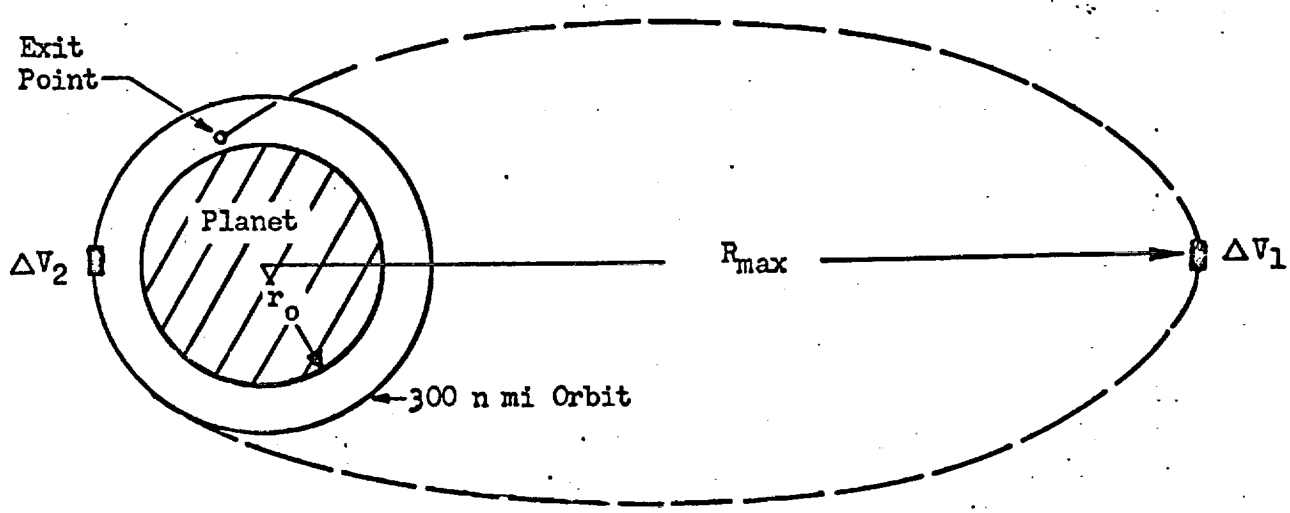
Three impulsive techniques (shown in Figure 5) for establishing the circular orbit following the graze were considered and evaluated with respect to their velocity requirements. For large exit velocities from the graze maneuver, a three impulse maneuver (Scheme 3) resulted in the lowest velocity requirements. For smaller exit velocities (less than escape velocity) a two impulse scheme was found to yield the lowest velocity requirements over a wide range of exit velocities. In this maneuver (Scheme 2) the vehicle coasts to apoapsis and is then brought down to orbit with two impulses, one applied at apoapsis and the other applied at the 300 n mi orbit altitude. The third maneuvering technique (Scheme 1) is a two impulse direct-to-orbit maneuver; the first impulse is applied at the exit altitude and the second at the 300 n mi altitude. A comparison of the impulsive velocity requirements for the three orbit establishment schemes is illustrated in Figure 6 .

An orbit establishment maneuver employing finite thrust was also analyzed and a comparison was made with the impulsive results. The finite thrust maneuver considered is similar to Scheme 1 in that the vehicle never exceeds the 300 n mi orbit altitude. In general the finite thrust maneuver required approximately 10 percent more ideal velocity than the comparable impulsive maneuver, Scheme 1. It was also found that the exit conditions (velocity magnitude and direction at the end of the graze maneuver) limit the minimum thrust level that can be used to perform the maneuver.

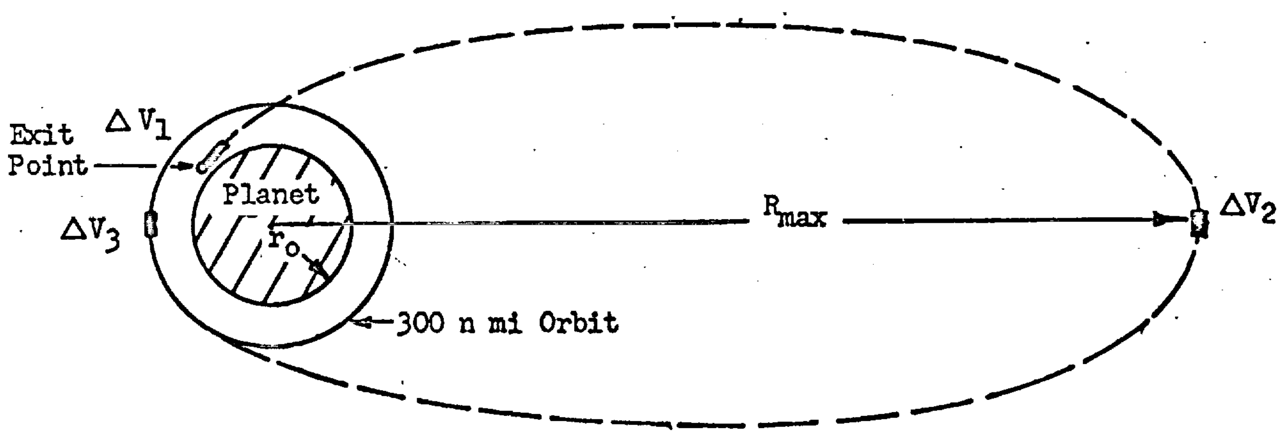
The selection of a particular orbit-establishment technique thus depends on the two factors: 1) the magnitude of the atmosphere graze exit velocity and 2) the ability of the vehicle to traverse the radiation belts. For vehicles that can pass through the radiation belts, Schemes 2 and 3 yield the lowest velocity requirements, Scheme 2 for exit velocities less than 33,000 ft/sec and Scheme 3 for exit velocities greater than 33,000 ft/sec. The velocity requirements determined for the impulsive analyses are adequate for systems with thrust-to-weight ratios above approximately 0.5. For lower thrust-to-weight ratios, the impulsive analysis tends to be optimistic. For vehicles which do not possess sufficient shielding for repeated penetration



Scheme 1



Scheme 2



Scheme 3

Figure 5 Impulsive Orbit Establishment Techniques

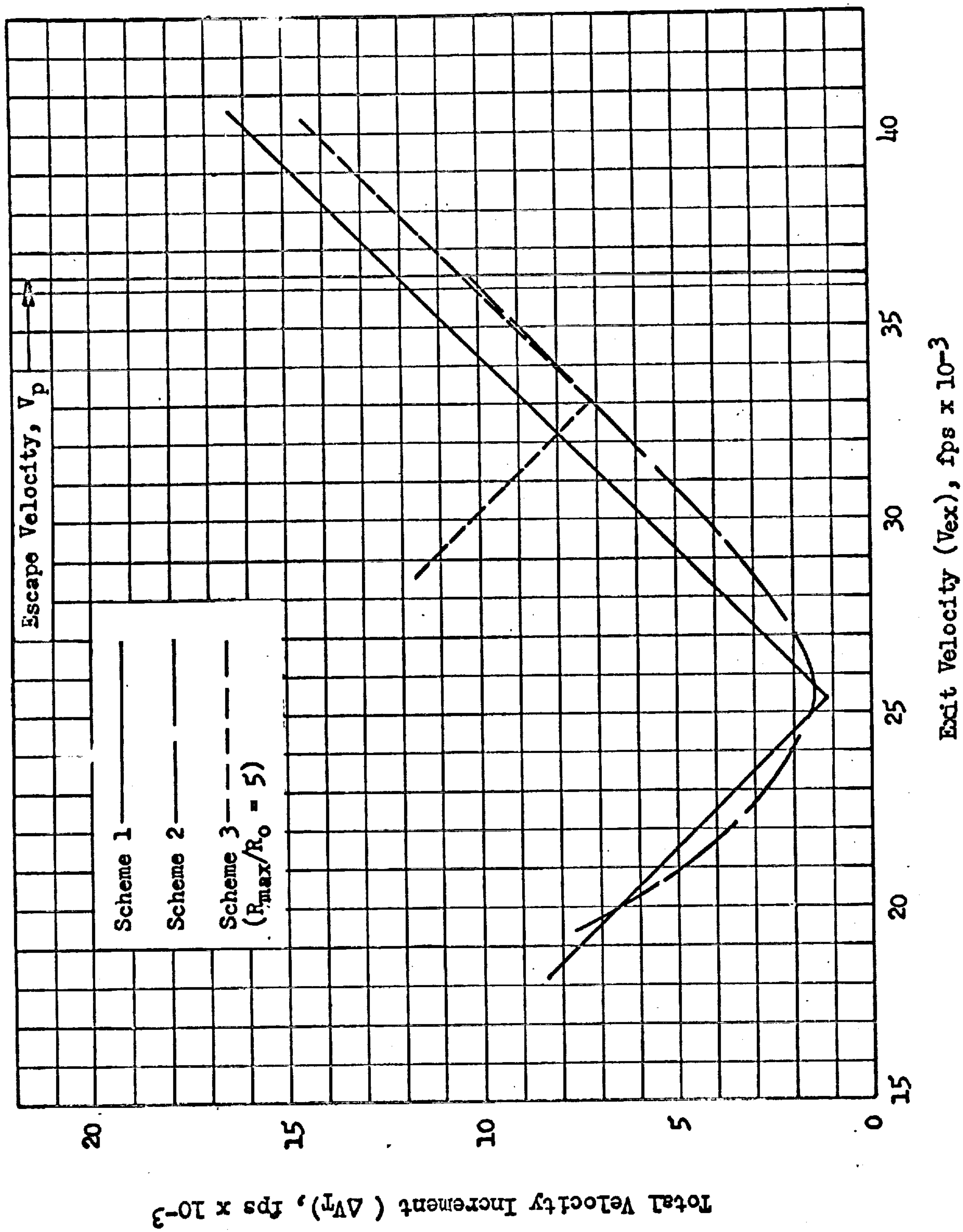


Fig. 6 Summary of Impulsive Velocity Requirements for Establishing a 300 n mi Orbit at Earth, γ_{ex} = 5 degrees

of the radiation belts, a direct-to-orbit maneuver should be used.

In general, the results indicate that the propulsion velocity requirement for orbit establishment after graze is approximately equal to the difference in graze exit velocity and orbital velocity. Thus if it is feasible, depending upon the vehicle design configuration and upon possible mission constraints, to utilize a ballistic or lifting vehicle, the graze maneuver can be used to reduce the propulsion required for orbit establishment compared to the propulsion for an entirely propulsive orbit-establishment maneuver.

PROPULSIVE/AERODYNAMIC BRAKING MANEUVER FOR EARTH RE-ENTRY

Deceleration of space vehicles approaching the Earth, Venus or Mars can be accomplished propulsively, aerodynamically, or by a combination of both methods. Because of the different rates at which propulsive and aerodynamic braking devices increase in weight as velocity increases, a system comprised of a propulsion system and an aerodynamic braking device may offer the lightest overall system.

The investigations presented were conducted to evaluate the applicability of propulsion/ablation systems to the braking maneuver for space vehicles approaching planetary atmospheres at extremely high velocities. It should be noted that the analyses presented were concerned only with weight considerations; thus, while certain assumptions might yield optimum designs having no propulsive braking, in practice, constraints such as maximum allowable g level can be stipulated such that propulsive braking of some specific magnitude must be applied prior to entry into the atmosphere. The questions of the existence of appropriate entry corridores at extremely high velocities and the ability to guide a vehicle to these corridors were evaluated in the analysis of terminal corrections, and thus were not considered in this investigation.

The results of the studies conducted indicate that for a given ablation shield characteristic, expressed as an exponential relationship between vehicle weight fraction for heat shield, and entry velocity, the optimum entry velocity is only moderately influenced by vehicle arrival velocity (i.e., velocity prior to propulsive or aerodynamic deceleration) and propulsion system characteristics. The optimum entry velocity (i.e., after propulsive deceleration) is, however, strongly influenced by the heat shield weight characteristic; available data on the nature of the characteristic are so widely variant that optimum entry velocities below 40,000 ft/sec and above 100,000 ft/sec were obtained.

The results of a parametric analysis are presented in Figure 7 . For any of the indicated results, the optimum propulsive ΔV is the difference between the arrival velocity corresponding to the particular mission and the optimum entry velocity associated with the applicable ablation characteristic. The best current data indicates an ablation characteristic between curves (3) and (4), and yields an optimum entry velocity of 56,000 ft/sec for the fast Mars-Earth trajectory considered (hyperbolic excess velocity of 48,000 ft/sec); the corresponding propulsive ΔV is slightly in excess of 4200 ft/sec. Thus for a "fast" return mission a small propulsion phase may be desirable; however for missions having lower Earth arrival velocities, aerodynamic deceleration alone results in maximum payload.

For the same Mars-Earth mission, an analysis of finite thrust systems indicated that optimum thrust-to-weight ratio (F/W) is dependent upon the entry velocity. At entry velocities greater or less than 46,000 ft/sec, the optimum F/W is less than 0.46 (it decreases rapidly as entry velocity increases above 46,000 ft/sec, and it decreases slowly as entry velocity decreases below 46,000 ft/sec). At the optimum entry velocity, 56,000 ft/sec, the optimum F/W is 0.32.

EARTH TERMINAL DECELERATION PHASE SYSTEMS

The inability of a parachute to decelerate a mass efficiently to very low velocity and the independence of a rocket device from any such constraint on operating regime suggests that a composite system for final deceleration of a landing vehicle might be more efficient than either device employed singly. The use of a rocket in conjunction with a parachute was therefore investigated for terminal phase deceleration during Earth landings. Additionally, a study of total systems comprised of parachute, retrorocket and impact device was conducted to optimize simultaneously the parachute terminal velocity and the vehicle impact velocity, and to indicate total system weights based on the selected subsystem weight values.

In the operation of these systems, the parachute is first deployed and slows the vehicle to terminal velocity (V_T). The vehicle continues to descend at terminal velocity to an altitude determined by the F/W of the retrorocket (the higher the F/W, the lower the ignition altitude). The retrorocket is then used to slow the vehicle to (surface) impact velocity (V_F). The value of V_F is generally restricted to below 25 ft/sec by touchdown stability or impact device design considerations, but if these restrictions are not present, then V_F is optimized along with V_T and F/W in the system analysis.

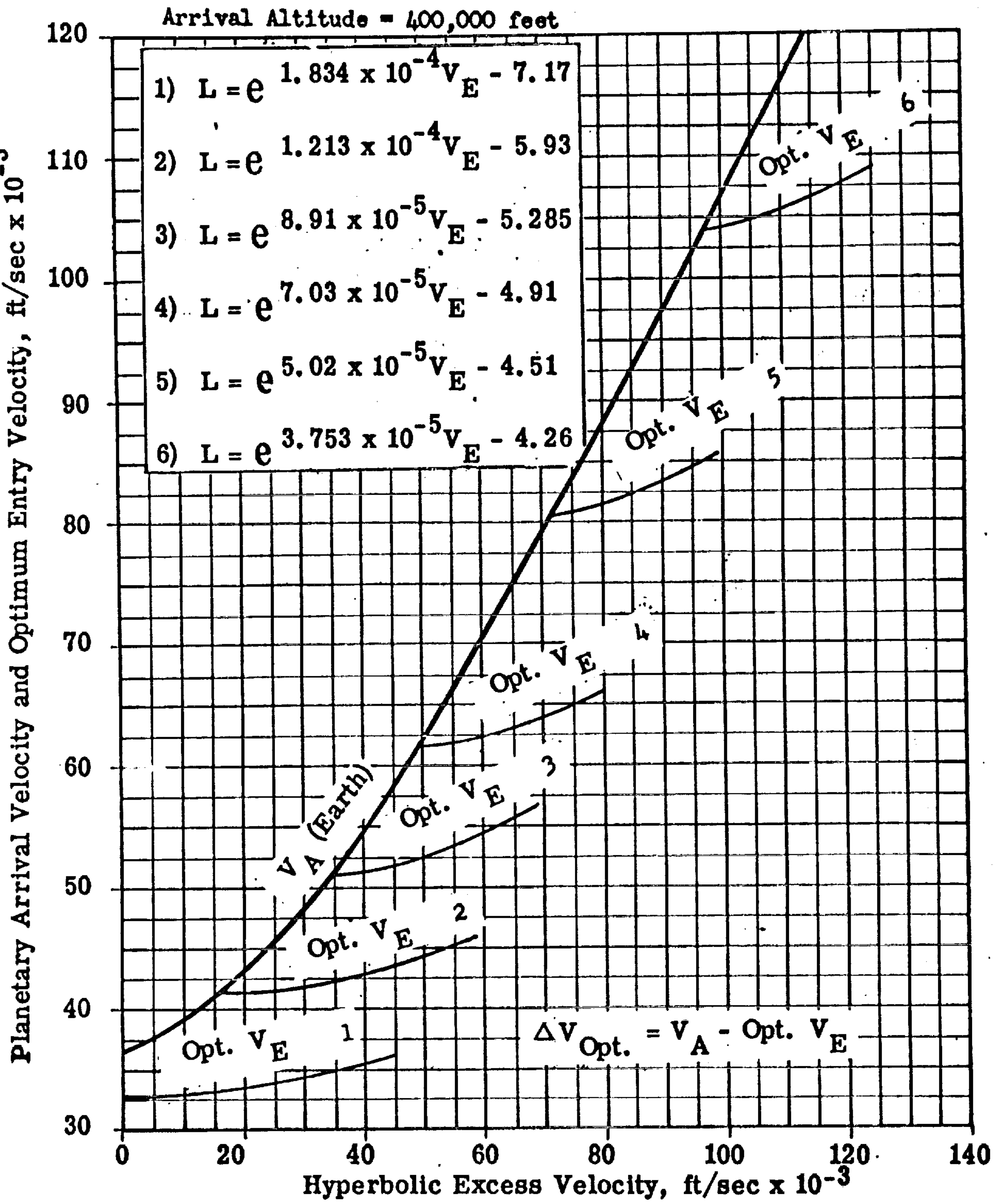


Fig.89 Optimum Propulsive ΔV (Impulsive) for Earth Re-Entry of Propulsive/Aerodynamic Systems.

The results of the investigation indicate that if a design impact velocity below 40 ft/sec is stipulated, the optimum landing system includes each of the three component parts; for a higher design impact velocity, the optimum system is composed of a parachute and impact device only. The minimum overall system weight resulted at a high design impact velocity (55 ft/sec for the nominal weight assumptions) where the parachute/impact device system is lighter than the pararocket/impact device system.

Results of the study are summarized in Tables 11 and 12. In Table 11, a nominal impact velocity is employed, and the characteristics of the optimum pararocket system are presented. Table 12 describes the characteristics of optimum systems for two nominal impact velocities (10 ft/sec and 25 ft/sec) and for systems in which impact velocity is an optimized parameter.

TABLE 11

EARTH PARAROCKET SYSTEM

(Design Impact Velocity = 10 ft/sec)

Optimum Thrust-to-Weight	Optimum V_T, ft/sec	Pararocket System Wt., Percent Gross Weight
1.5	70	3.9

TABLE 12

COMBINED SYSTEM WEIGHTS

Minimum Weight Configuration	Optimum V_F = 55 ft/sec	V_F = 10 ft/sec	V_F = 25 ft/sec
	Parachute/Impact Device	Pararocket/ Impact Device	Pararocket/ Impact Device
Percent Gross Wt.	3.1	4.4	4.1

The resultant system weights presented in Tables 11 and 12 are dependent on the nominal weight values assigned to each of the system components. Variations in parachute and rocket weights were analyzed to evaluate their effects on optimum system configuration and weight. A 50 percent increase in parachute weight for the pararocket/impact device system increased the optimum impact velocity 5 ft/sec to 60 ft/sec and increased the system weight to a total of approximately 3.5 percent. A 50 percent increase in impact device weight reduced the optimum impact velocity by about the same amount and the total system weight increased to approximately 3.8 percent.

Preliminary error analysis of the retrorocket indicates that the most critical conditions are low thrust level operation or late ignition. Both these conditions result in impact velocities considerably above the expected value. Thus in the actual design, the capability of the impact energy absorbing device would have to include a margin for errors in rocket operation.

EARTH - MARS MISSIONS

MARS TRAJECTORY SELECTION

The propulsion requirements for a round trip mission to Mars are strongly dependent upon the launch dates and transfer durations selected for the outbound and return legs of the journey.

A variety of Earth-Mars trips, characterized as slow or fast were examined. The hyperbolic velocities vary cyclically, and repeat (except for a change on the order of 1000 ft/sec from one period to the next, caused by the eccentricity and inclination of the Mars orbital plane) each synodic period, or 780 days.

Selection of a representative mission profile for a Mars round trip was predicated on the restrictions that 1) Earth-departure and Mars-departure velocity requirements should be minimized and 2) that the trip time and the Mars-departure propulsion should be reduced at the expense of Earth departure propulsion. The former stipulation is based on recognition of the fact that aerodynamic deceleration will be used for landing at both Mars and Earth; except for extremely high entry velocities or extremely poor aerodynamic heat shields, propulsion systems are heavier than ablative shields. The latter criterion takes into account the fact that an Earth-departure propulsion system, because it has little or no requirement for shielding or insulation, is a far more efficient device (i.e., has a superior propellant fraction) than is a Mars-departure propulsion system. Thus, the bias indicated by condition (2) is beneficial to the overall vehicle system.

A selected mission is described in Table 13 . Although vehicle design data are not currently known with sufficient accuracy to assure that this profile is optimum, there is reasonable certainty that a profile yielding major reductions in vehicle requirements is not likely.

TERMINAL CORRECTIONS FOR EARTH-MARS TRAJECTORIES

Landing on Mars can be performed with the vehicle entering an orbit about Mars and then descending wholly or in part to the surface, or the space vehicle can employ atmospheric braking for direct descent to the surface. The first concept could be of value for early missions where it was not deemed possible to rely on a direct aerodynamic entry without surveillance, equipment checkout, etc., and the second for later, maximum payload missions. In either case there is a requirement for terminal corrections, since mid-course correction analyses for the missions studied have shown that the vehicle approach trajectory accuracy is inadequate. The terminal correction propulsion requirements for both landing concepts were analyzed.

TABLE 13

SELECTED MARS MISSION TRAJECTORIES

	Earth-Mars	Mars-Earth
Launch Date		
Gregorian	6 June 1971	6 Sept. 1971
Julian	2441108.5	2441200.5
Transfer Time, days	80	260
Hyperbolic Departure Velocity, ft/sec	20,500	18,000
Hyperbolic Arrival Velocity, ft/sec	34,400	27,000
Mars Stay-time, days	12	
Total Mission Time, days		352

TABLE 14

EARTH-MARS MISSIONS

Launch Date	Transfer Time, days	Hyperbolic Arrival Velocity, ft/sec	Nominal Asymptotic Approach Distance, n mi	Actual Asymptotic Approach Distance, n mi
For Orbit-Establishment Mission				
6 Dec. 1964	250	12,000	3,400	5850
For Atmospheric Entry Missions				
19 May 1971	170	11,000	3,190	6,650
6 June 1971	80	34,400	2,025	2,500

For missions which utilize a propulsion phase to establish a 300-n mi circular orbit, a study has been performed in conjunction with Reference 1. A relatively long transfer mission (Table 14) with a low arrival velocity was selected as representative of propulsive orbit-establishment missions since faster missions result in excessive propulsion ΔV and propulsion system weight requirements. The terminal correction velocity requirements as a function of range were determined for this mission.

An analysis of the effects of errors in executing the terminal correction on the final orbit was then performed. A tolerance of 10-percent (30 n mi) in the (apoapsis) altitude of the orbit was selected for determining the range for performing the terminal correction maneuver and the required terminal correction velocity increment.

Terminal corrections required for atmospheric braking missions were investigated for the selected Earth-Mars missions and for a typical "slow" mission; the method was similar to that used in the analysis of Earth terminal corrections. The entry corridor limits for a ballistic drag vehicle were used.

The required terminal corrections for the missions are summarized in Table 15. The magnitudes of the required velocity increments are sufficiently small to preclude the need for dual correction schemes as were used for Earth terminal corrections. The velocity requirements were determined for propulsion systems with a nominal 0.3 initial F/W ratio. However, the velocity requirements analysis indicated that for a F/W range from 0.1 to 0.5, the change in results is negligible. The results indicate that for aerodynamic direct-landing maneuvers, the use of terminal correction will permit successful entry into the entry corridor and a propulsive phase for deceleration is not required.

TABLE 15

NOMINAL CORRECTIONS FOR MARS MISSIONS

Mission	Hyperbolic Arrival Velocity, ft/sec	Deviations from Nominal Asymptotic Miss Distance, n mi	Correction Range, n mi	Correction Velocity Increment, ft/sec
Orbit Establishment	12,000	2450	46,000	630
Direct Entry	11,000	2460	72,000	380
Direct Entry	34,400	475	44,000	360

PROPULSIVE MARS ORBIT ESTABLISHMENT
AND DEPARTURE MANEUVERS

The propulsion requirements for Mars orbit establishment and departure maneuvers were determined using computer-simulated trajectories. Representative vehicle and propulsion system characteristics were employed to determine optimum values of thrust-to-weight ratio for these maneuvers. The effects of engine weight, propellant tank weight and specific impulse on optimum thrust-to-weight ratio, and the region in which thrust-to-weight ratio has a small effect on payload were evaluated. Those results are presented in Figure 8 . The selected value of hyperbolic excess velocity (V_H) is representative of a minimum-energy Mars mission; although V_H affects payload, it has little effect on the selection of optimum thrust-to-weight ratio.

MARS ORBIT ESTABLISHMENT FOLLOWING
AN ATMOSPHERIC GRAZE

The impulsive velocity requirements for establishing a 300-n mi orbit at Mars following an atmospheric graze maneuver were evaluated in a similar fashion to those for Earth. The total impulsive velocity requirement for establishing orbit are presented in Figure 9 as a function of the magnitude of velocity existing at the end of the graze maneuver. The analysis was performed for three maneuvering schemes which are described in the Earth analysis (see Figure 5).

A trend similar to that noted in the Earth analysis was obtained: schemes 2 and 3 yield the lowest velocity requirements over most of the range of exit velocities, Scheme 2 for exit velocities less than 14,000 ft/sec, Scheme 3 for exit velocities greater than 14,000 ft/sec.

The applicability of the atmospheric graze maneuver to a given mission depends on the vehicle configuration and constraints (i.e., maximum g limit, heat shielding, etc.). For vehicles capable of executing the graze maneuver, considerable saving in propulsive energy can be realized over a direct orbit establishment maneuver.

PROPULSIVE/AERODYNAMIC BRAKING
MANEUVER FOR MARS ENTRY

Propulsive/Aerodynamic braking systems for landing on Mars are similar to systems described previously for Earth re-entry. The major problem in this case, as before, is the accurate definition of the ablation shield weight; as a result, the variety of ablation characteristics utilized for Earth re-entry vehicle analysis was employed for parametric study of Mars entry.

Nominal Vehicle: $I_s = 420$ sec, $K_T = 0.16$,
$K_E = 0.05$, $V_h = 12{,}000$ ft/sec

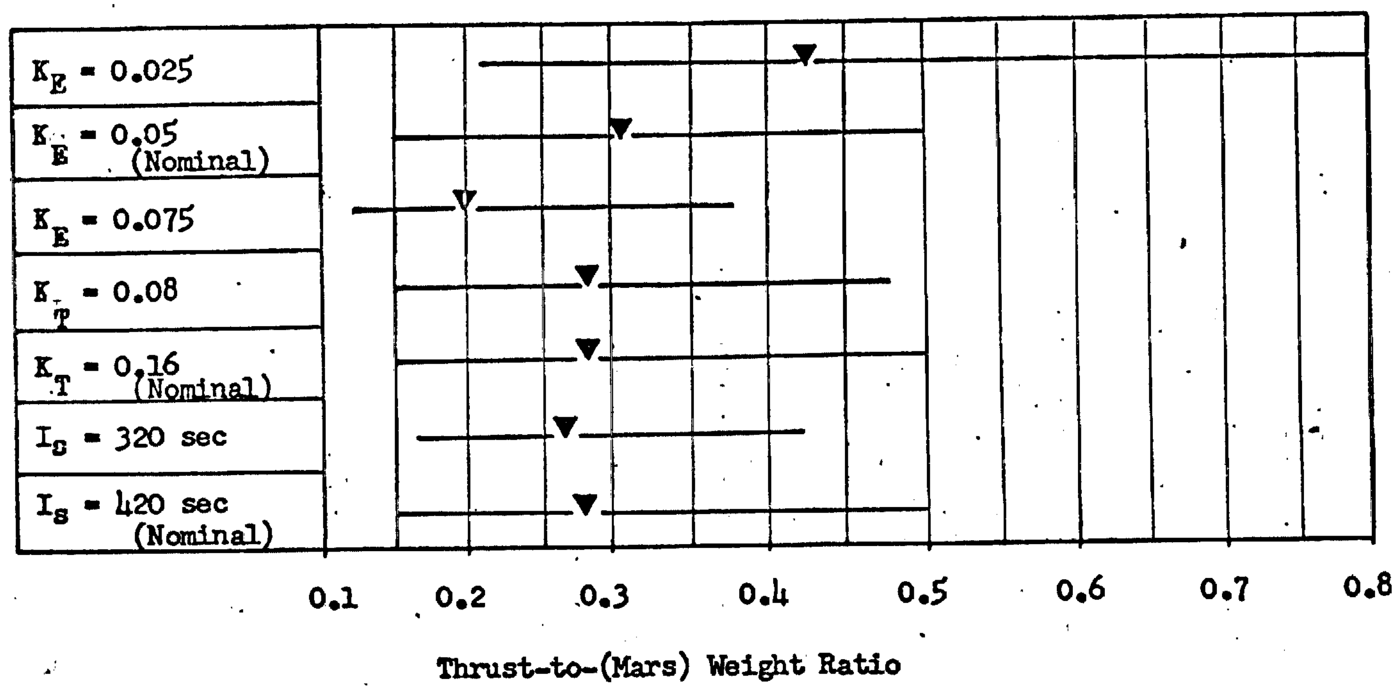


Figure 8 Thrust-to-Weight Ratio Variation for a One-Percent Change in Payload for Mars Orbit Establishment or Departure Maneuver.

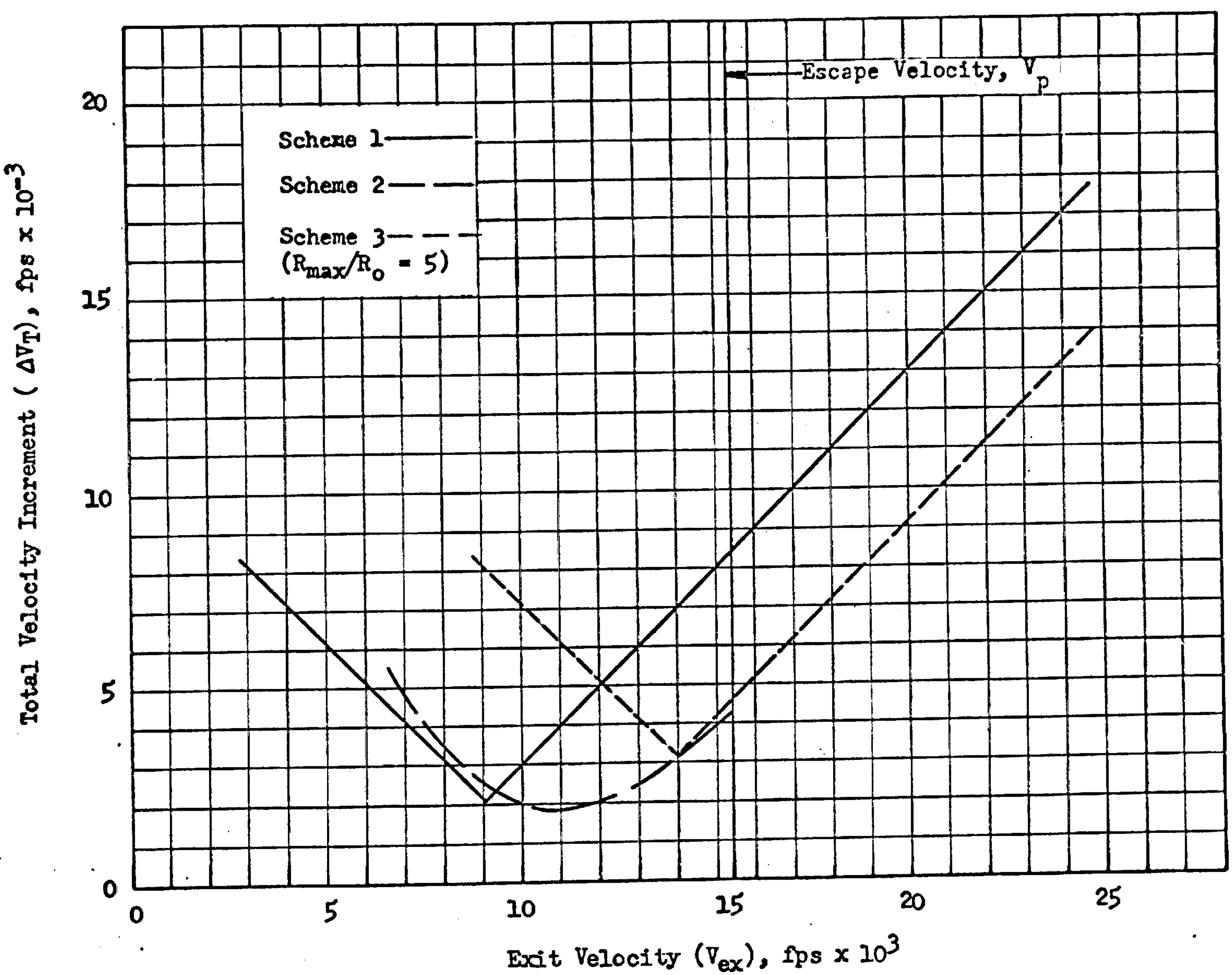


Fig. 9 Summary of Impulsive Velocity Requirements for Establishing a 300 n mi Orbit at Mars, γ_{ex} = 20 degrees

Results of the study are presented in Figure 10. For a selected value of hyperbolic excess velocity and a particular ablation characteristic, the optimum propulsive ΔV is found by measuring the difference between the arrival velocity and the applicable optimum entry velocity. The results are similar to those for Earth in that a propulsive ΔV is useful to achieve maximum payload only for extremely high arrival velocities.

MARS TERMINAL DECELERATION PHASE SYSTEMS

A study of parachute/retrorocket/impact device systems for the terminal deceleration phase of a Mars landing was conducted to determine the optimum parameters (parachute terminal velocity, rocket F/W, impact velocity) and the system weight. The investigation was similar to that conducted previously for Earth landings. The lower density of the Mars atmosphere near the surface results in greater parachute weights (compared to Earth values) to achieve low terminal velocities. Thus, the results of the investigation indicate that for design impact velocities below 75 ft/sec, the optimum landing system includes each of the three component parts. The design impact velocity must be higher than 75 ft/sec for the optimum system to be composed of only the parachute and impact device. The lightest overall system obtained was a pararocket/impact device combination with an impact velocity of 35 ft/sec, a rocket F/W of 0.8, and a parachute terminal velocity of 120 ft/sec; the system constituted 4.7 percent of the landed weight.

A summary of the minimum-weight systems for the optimum impact velocity, and for impact velocities of 10 ft/sec and 25 ft/sec, is presented in Table 16.

TABLE 16

COMBINED SYSTEM WEIGHTS

	Optimum V_F = 35 ft/sec	V_F = 10 ft/sec	V_F = 25 ft/sec
Minimum Weight Configuration	Pararocket/Impact Device	Pararocket/ Impact Device	Pararocket/ Impact Device
Percent Gross Weight	4.7	5.0	4.8

MARS PROPULSIVE TAKEOFF AND LANDING

Advanced planetary missions include landings on, and takeoffs from, the planet Mars. The takeoffs must be propulsive maneuvers. Although the propulsive/aerodynamic braking analysis has shown that aerodynamic deceleration is, in general, more efficient, early missions may, because of atmospheric uncertainties or mission philosophy, use a propulsive landing.

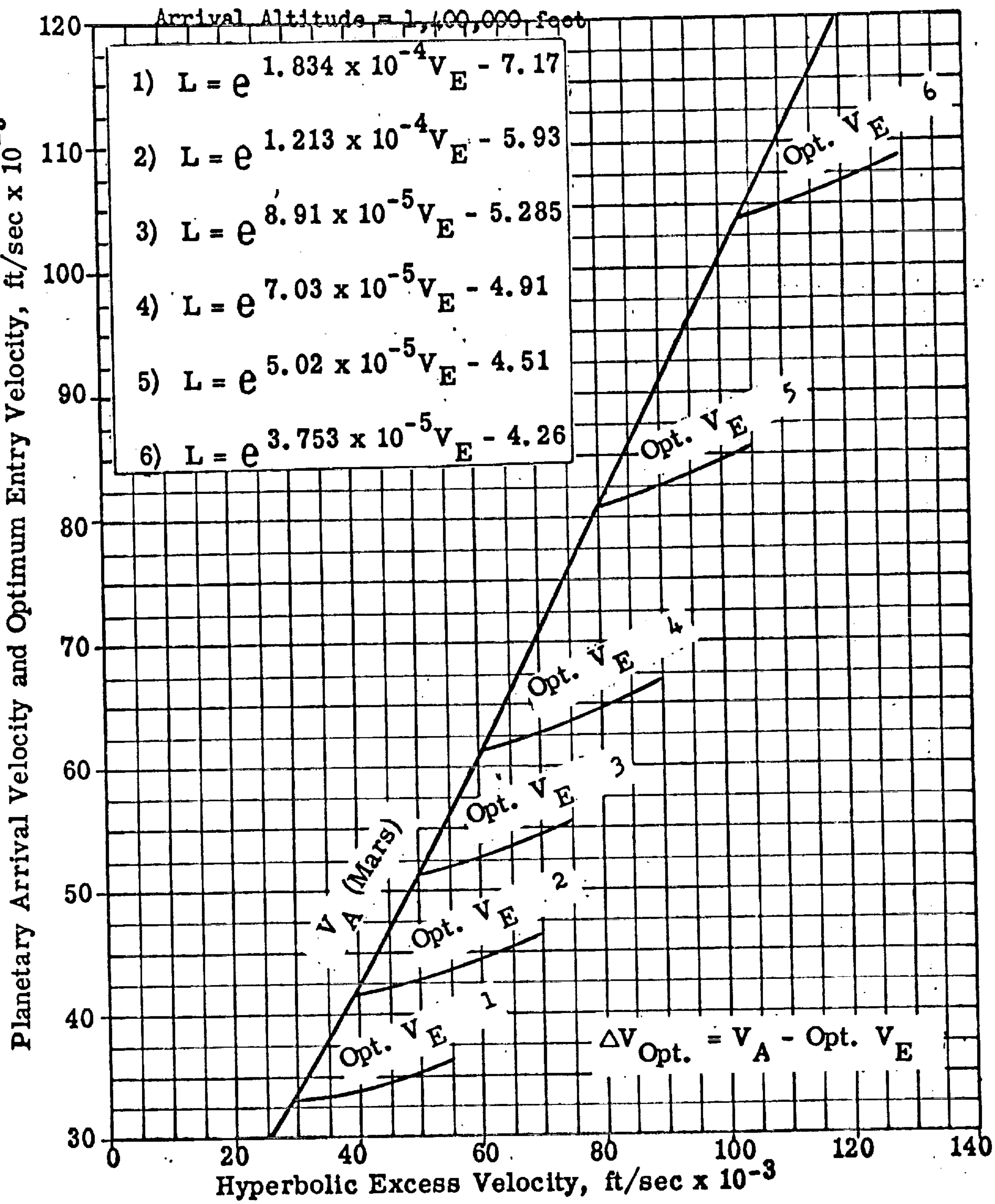


Fig. 10 Optimum Propulsive ΔV (Impulsive) for Mars Entry of Propulsive/Aerodynamic Systems.

An investigation of propulsive landing takeoff and maneuvers at Mars was performed to determine the ideal velocity requirements for these maneuvers and to evaluate the optimum thrust-to-weight ratio based on representative values of vehicle and propulsion system characteristics. Velocity requirements were determined for descent from a 50-n mi circular orbit and for ascent to a 300-n mi circular orbit. Equatorial orbits in the same direction as planet rotation were utilized for both maneuvers. Because of the relatively low velocity requirements, only single stage vehicles were considered. The propulsion system employed in both instances was a O_2/H_2 pump-fed system whose characteristics are listed in Table 17.

TABLE 17

ENGINE PERFORMANCE

	Mars Takeoff and Landing Engine
Chamber Pressure, psia	650
Nozzle Expansion Area Ratio	50:1
Vacuum Specific Impulse, seconds	432
Surface Specific Impulse, seconds	411

The results indicate that a thrust-to-Mars weight ratio of 2.0 is approximately optimum for a Mars takeoff; the corresponding ideal velocity requirement is approximately 14,200 ft/sec. For the Mars landing, an integrated trajectory was determined only for an initial thrust-to-(Mars) weight ratio of 0.855, resulting in a thrust-to-(Mars) weight of about 2.0 at touchdown. The landing maneuver required an ideal velocity increment of about 12,400 ft/sec the difference (i.e., aside from the inherent difference that is characteristic of landing and takeoff ΔV requirements) partly reflects the difference in orbit heights for the two maneuvers, and partly reflects the fact that drag is beneficial to a landing maneuver but detrimental to a takeoff maneuver.

EARTH-VENUS MISSIONS

TRAJECTORY SELECTION

The trajectory characteristics of a variety of Earth-Venus missions were examined. The indicated missions are repeated every 584 days because the orbit of Venus is very nearly circular (eccentricity of 0.007 as compared to Mars-orbit eccentricity of 0.093); the cycle-to-cycle repeatability of Venus missions is close to exact.

Selection of a mission profile for a Venus round trip for reference use was governed by the same criteria as those mentioned earlier for the Mars trip. The primary objective was to provide a relatively low Venus-departure velocity requirement (since it is a propulsive maneuver, and the propellant to execute it must be stored in space) even at the expense of somewhat higher Earth-departure, Venus-arrival and Earth arrival velocities. The nominal mission is described in Table 18 .

TABLE 18

SELECTED VENUS MISSION TRAJECTORIES

	Earth-Venus		Venus-Earth	
Launch Date				
Gregorian	24 June 1965		31 Dec. 1965	
Julian	2438935.5		2439125.5	
Transfer Time, days	185		110	
Hyperbolic Departure Velocity, ft/sec	27,000		19,500	
Hyperbolic Arrival Velocity, ft/sec	29,000		10,000	
Venus Stay-Time, days		5		
Total Mission-Time, days				300

TERMINAL CORRECTIONS FOR EARTH-VENUS TRAJECTORIES

Landing concepts include first entering an orbit and then descending to the surface, or direct atmospheric braking to the surface. Terminal corrections are required for either method since midcourse correction analyses have shown that the vehicle will not approach the planet within allowable accuracy tolerances.

The propulsion requirements for the terminal corrections have been evaluated in a manner similar to the Earth and Mars studies. An analysis of terminal corrections required to facilitate propulsive establishment of a 300-n mi circular orbit has been performed in conjunction with Reference 1 . The Earth-Venus trajectory used was a relatively long transfer mission having a low hyperbolic arrival velocity, since faster missions usually result in excessive propulsive ΔV and propulsion system weight requirements. Analysis of terminal correction errors was performed to evaluate deviations in altitude of the propulsively-established orbit. The errors encountered were range-dependent, and therefore their effects were evaluated as a function of range. An allowable tolerance of 10 percent (30 n mi) in the deviation of apoapsis altitude of the orbit was selected for determining the appropriate range for applying the terminal correction.

Terminal corrections required for direct aerodynamic entry into the Venusian atmosphere were analysed in a manner similar to previous Earth and Mars studies. The velocity requirements for a single terminal correction were considered excessive; therefore, the use of two corrections was investigated. In the dual correction scheme, the first correction was applied at 100,000 n mi. A second correction range was determined to give an entry angle deviation which satisfied the entry corridor requirement. The magnitude of the velocity requirements for the terminal corrections were determined for propulsion systems with a nominal 0.3 initial F/W ratio; however, the velocity requirements analysis indicated that for a F/W range from 0.1 to 0.5, the change in results is negligible.

The nominal trajectory conditions and required corrections for the selected orbit establishment and aerodynamic entry missions are summarized in Table 19.

TABLE 19

SUMMARY OF TERMINAL CORRECTIONS FOR VENUS MISSIONS

Mission	Number of Terminal Corrections	Hyperbolic Arrival Velocity, ft/sec	Deviation from Nominal Asymptotic Miss Distance, n mi	Correction Range, n mi	Correction Velocity Increment, ft/sec	Total Terminal Correction ΔV, ft/sec
Orbit Estab-lisment	1	12,500	2,850	61,000	550	550
Direct Entry	2	15,900	3,150	First 100,000	490	
				Second 16,000	130	620

The dual correction scheme, used for the direct entry mission, reduced the velocity increment approximately 3200 ft/sec over that required for a single correction to achieve the desired entry conditions. This difference clearly warrants use of a dual correction scheme despite the addition of a requirement for engine restart capability. With the terminal corrections, the required entry corridor can be successfully established, as in the case of Earth and Mars, without an additional propulsive deceleration phase.

PROPULSIVE VENUS ORBIT ESTABLISHMENT AND DEPARTURE MANEUVERS

A study was conducted to determine optimum values of thrust-to-weight ratio for Venus orbit-establishment and departure maneuvers. Representative values of typical cryogenic and noncryogenic propulsion system characteristics were utilized. A single mission hyperbolic velocity was selected; this was demonstrated in previous studies to affect payload but not optimum F/W selection.

The results are summarized in Table 20 . The optimum values of F/W, expressed in terms of local weight, are similar to the optimum thrust-to-local weight values obtained in analyses of similar maneuvers at other planets.

TABLE 20

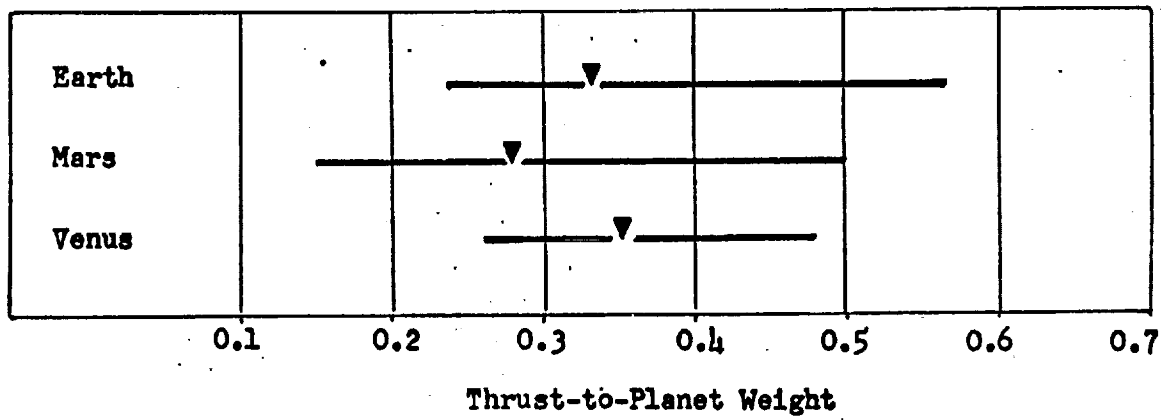
OPTIMUM THRUST-TO-WEIGHT RATIO

	Nonredundant System (K_E = 0.025)	Redundant System (K_E = 0.050)
Cryogenic System	0.40	0.35
Noncryogenic System	0.40	0.28

The results, considered in conjunction with results for other planets, indicate a consistent optimum thrust-to-planet-weight ratio for orbit establishment maneuvers. This is demonstrated in Figure 11 ; corresponding values of thrust-to-Earth-weight ratio are presented for comparison.

VENUS ORBIT ESTABLISHMENT FOLLOWING AN ATMOSPHERIC GRAZE

The impulsive velocity requirements for establishing a 300-n mi orbit at Venus following an atmospheric graze maneuver were evaluated in a similar fashion to those for Earth. The total impulsive velocity requirement for



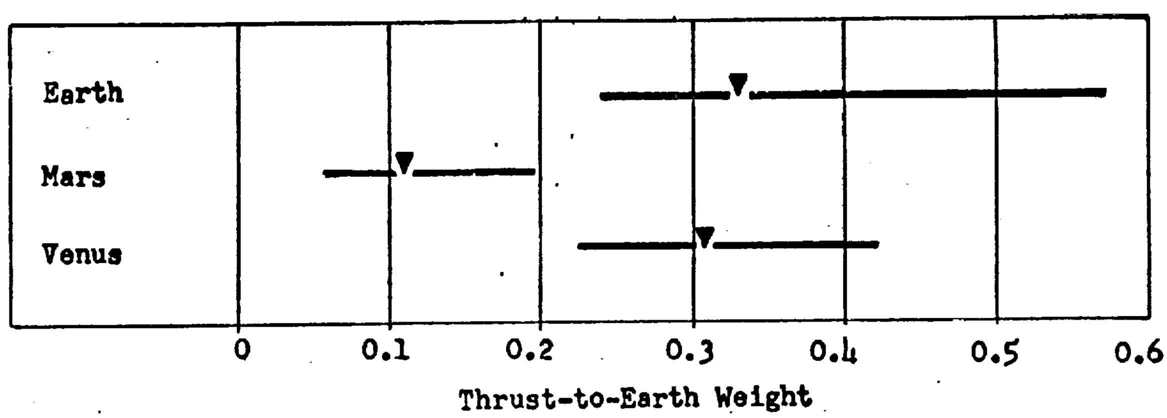


Fig. 11 Thrust-to-Weight Comparison for Nominal Planetary Vehicles

establishing orbit are presented in Figure 12 as a function of the magnitude of velocity existing at the end of the graze maneuver. The analysis was performed for three maneuvering schemes which are described in the Earth analysis. A trend similar to that noted in the Earth analysis is indicated in that Schemes 2 and 3 yield the lowest velocity requirements over most of the range of exit velocities, Scheme 2 for exit velocities less than 30,000 ft/sec and Scheme 3 for exit velocities greater than 30,000 ft/sec.

For vehicles capable of executing a graze maneuver, a propulsion savings, as in the case of Earth and Mars, can be achieved.

PROPULSIVE/AERODYNAMIC BRAKING MANEUVER FOR VENUS ENTRY

Propulsive/Aerodynamic braking systems for landing on Venus are similar to systems described previously for Earth re-entry. The major problem in this case, as before, is the accurate definition of the ablation shield weight; as a result, the variety of ablation characteristics utilized for Earth re-entry vehicle analysis was employed for parametric study of Venus entry.

Results of the study are presented in Figure 13 . For a selected value of hyperbolic excess velocity and a particular ablation characteristic, the optimum propulsive ΔV is found by measuring the difference between the arrival velocity and the applicable optimum entry velocity.

VENUS TERMINAL DECELERATION PHASE SYSTEMS

A study of parachute/retrorocket/impact device systems for the terminal deceleration phase of a Venus landing was conducted to determine the optimum parameters (parachute terminal velocity, rocket F/W, impact velocity) and the system weight. The investigation was similar to that conducted previously for Earth landings. The high density of the atmosphere at the surface of Venus suggests that for parachute/retrorocket/impact device systems, the optimum parachute terminal velocity will be substantially lower than it is for Earth or Mars landing systems.

The results indicate that the pararocket/frangible-tube impact device system is lighter for design impact velocities up to 25 ft/sec; for a higher design impact velocity, the parachute/frangible-tube impact device system is lighter. The minimum-weight system has an impact velocity of approximately 40 ft/sec, uses the parachute and frangible-tube system, and has an approximate system weight of 1.8 percent of the landing vehicle gross weight.

A summary of the minimum-weight systems for the optimum impact velocity, and for impact velocities (V_F) of 10 ft/sec and 25 ft/sec, is presented in Table 21 .

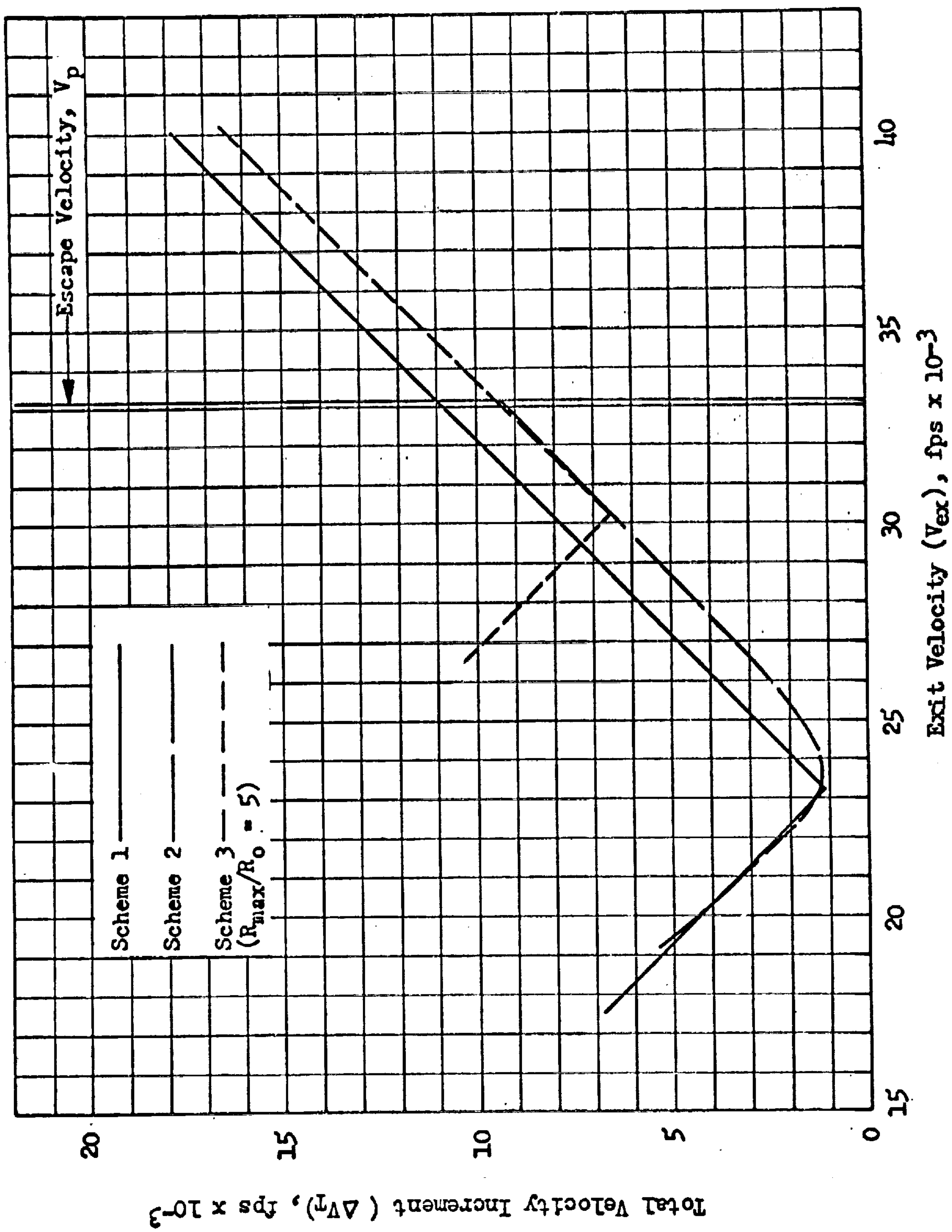


Fig. 12 Summary of Impulsive Velocity Requirements for Establishing a 300 n mi Orbit at Venus, γ_{ex} = 5 degrees

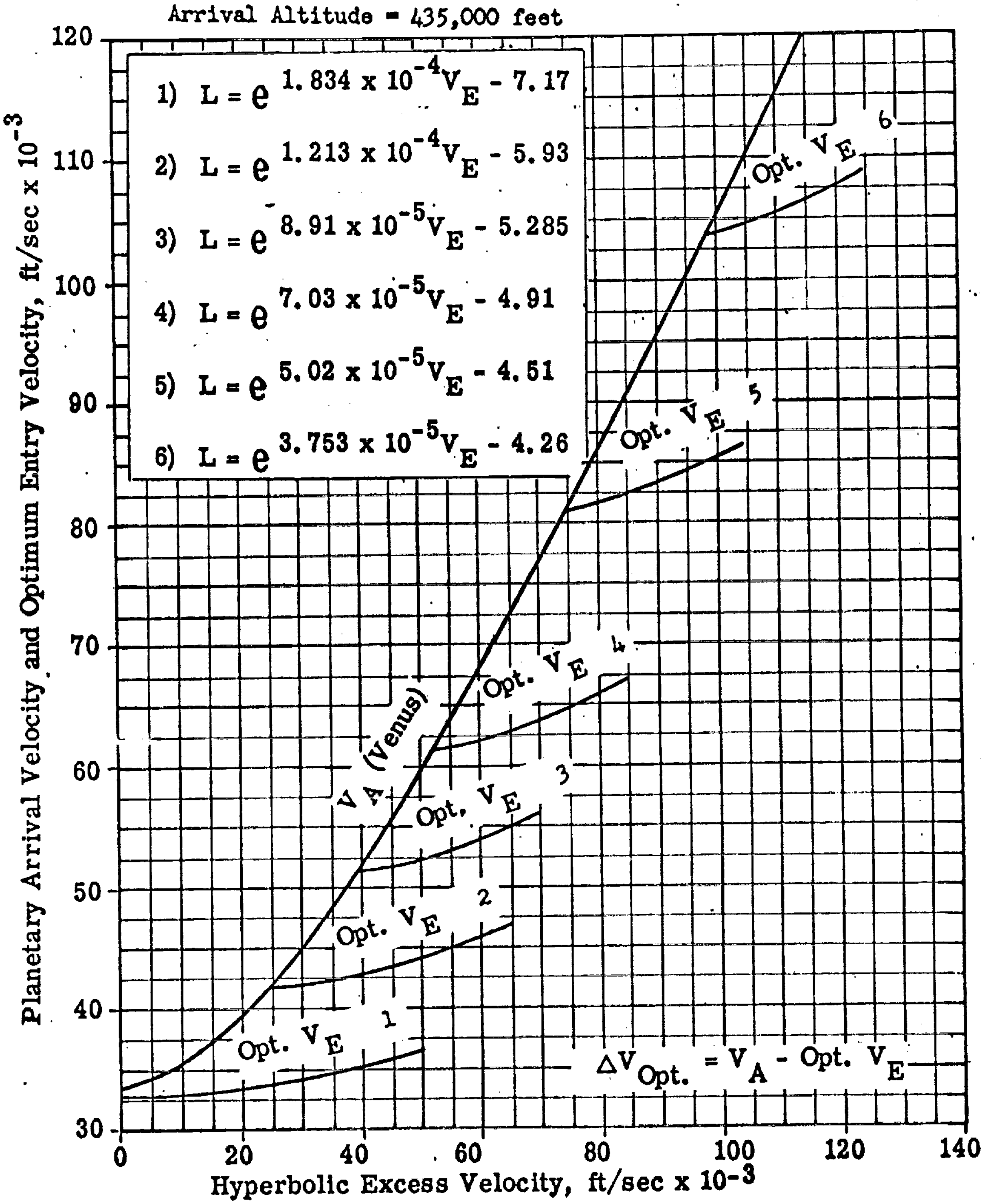


Fig. 13. Optimum Propulsive ΔV (Impulsive) for Venus Entry of Propulsive/Aerodynamic Systems.

TABLE 21

COMBINED SYSTEM WEIGHTS

System	Configuration	Impact Velocity (V_F, ft/sec)	Percent Gross Weight
Minimum Weight	Parachute/Impact Device	42	1.8
Limited V_F	Pararocket/Impact Device	10	3.2
Limited V_F	Pararocket/Impact Device	25	2.7
Limited V_F	Parachute/Impact Device	10	5.0
Limited V_F	Parachute/Impact Device	25	2.7

VENUS TAKEOFF PROPULSION REQUIREMENTS

Takeoff from the planet Venus is a propulsive maneuver made particularly difficult by the high drag resistance and poor rocket performance experienced at low altitudes in the dense Venusian atmosphere. Integrated trajectories for Venus takeoff to a 300-n mi circular planetary orbit were computed to determine the propulsion requirements for performing the takeoff maneuver necessary for round trip missions. Because of the high ideal velocity requirement of the mission, 2-, 3-, and 4- stage vehicles were utilized for the Venus takeoff. First stage thrust-to-(Venus) weight ratios of 1.3 to 1.7 were considered, and stage propellant fractions were assumed in all cases to be 0.9.

The characteristics of the engine systems used in this study are presented in Table 22 . The engine systems considered are pump-fed designs using O_2/H_2 propellants.

TABLE 22

ENGINE PERFORMANCE

	Venus Takeoff Engines		
	First Stage	Upper Stages	
Chamber Pressure, psia	1000	1000	1000
Nozzle Expansion Area Ratio	5:1	50:1	10:1
Vacuum Specific Impulse, seconds	381	435	403
Surface Specific Impulse, seconds	310	-277	206

The 50:1 expansion area ratio engine was used in the second stage of the 2-stage vehicles, the third stage of the 3-stage vehicles, and the third and fourth stages of the 4-stage vehicles. The 10:1 expansion area ratio engine was used as a second stage engine in both the 3 and 4-stage vehicles.

The results for the analysis of Venus takeoff-to-300-n mi orbit are presented in Table 23 . This table shows the thrust-to-weight ratio of each stage and the corresponding ideal velocity requirement necessary for mission accomplishment. The payload which would result if each stage had a propellant fraction of 0.9 is also presented. The fact that the 4-stage vehicle has the lowest ideal velocity requirement of the vehicles considered indicates that a throttleable engine (operated regressively until the vehicle passes above the dense portion of the atmosphere) might be best suited to the Venus takeoff mission.

TABLE 23

VENUS TAKE-OFF VEHICLE PERFORMANCE

Gross Weight = 100,000

	Thrust-to-Venus-Weight 1st Stage	2nd Stage	3rd Stage	4th Stage	Mission Ideal Velocity Requirement - ft/sec	Payload - LB ($\lambda_{P_{all\ stages}} = 0.90$)
2 Stages	F/W = 1.3	F/W = 0.77	---	---	39,800	1510
	F/W = 1.3	F/W = 1.30	---	---	39,000	1670
	F/W = 1.3	F/W = 2.20	---	---	39,200	1620
	F/W = 1.5	F/W = 0.87	---	---	40,100	1460
	F/W = 1.5	F/W = 1.50	---	---	39,800	1510
	F/W = 1.5	F/W = 2.49	---	---	40,100	1450
3 Stages	F/W = 1.3	F/W = 1.3	F/W = 1.3	---	40,000	2170
	F/W = 1.5	F/W = 1.5	F/W = 1.5	---	39,700	2250
	F/W = 1.7	F/W = 1.7	F/W = 1.7	---	40,000	2180
4 Stages	F/W = 1.5	F/W = 1.5	F/W = 1.5	F/W = 1.5	38,800	2940

LUNAR MISSIONS

INITIAL AND MIDCOURSE MANEUVERS

Trajectory phases preceding the propulsive lunar landing maneuvers were analyzed to determine favorable characteristics of overall round trip lunar missions, and thereby establish appropriate initial conditions for landing investigations. The selection of a mission profile is governed by the trade-off between propulsion requirements (for Earth departure and lunar orbit establishment; these are minimum for an approximate 5-day transfer) and shielding and life support, which favor shorter trips.

For transfer times shorter than 2.0 days, the Earth-phase velocity increases rapidly while for times longer than 2.0 days, the Earth-phase velocity is practically constant. The velocity of the vehicle, and therefore the lunar-phase velocity requirement as it enters the lunar gravity field, increases rapidly with the shorter transfer-time trajectories.

Preliminary analysis was conducted to examine the trip time with respect to shielding and life support equipment, and the results indicate that shield requirements cannot be defined with sufficient clarity to provide a precise value of optimum trip time. Review of available shield and life support information (see, for example, Reference 2), together with the propulsion requirements, indicates trips in the 2- to 3-day range are suitable for lunar missions.

The lunar mission differs from interplanetary missions in that the transit time is approximately three days contrasting to transit times of a hundred days or more for other space missions. Errors existing in the booster guidance and propulsion system would cause the vehicle to miss its rendezvous point at the moon by several thousand miles. Numerous midcourse analyses have been performed in connection with programs such as Apollo, Surveyor and Ranger. Similar analyses, conducted at Rocketdyne under NASA contract NAS 7-88, Space Transfer Phase Propulsion Systems, are described in Reference 1 . These analyses have determined that a midcourse correction scheme employing three maneuvers is satisfactory. Based on typical injection errors of 1 n mi in position and 10 ft/sec in velocity and including errors in midcourse position, guidance and execution accuracy, the results have shown that the total midcourse velocity requirements for a 0.99 probability of success are less than 200 ft/sec while the rms error existing at the aim point is less than 5 n mi in position and 0.5 ft/sec in velocity. For a mission which includes a propulsive phase to establish a lunar orbit, this accuracy is sufficient; however, for missions that involve circumnavigating the moon, further corrections will most likely be required to improve the trajectory accuracy.

LANDING AND TAKEOFF TRAJECTORY CONCEPTS

A vehicle on an Earth-moon coast trajectory approaches the moon along a selenocentric hyperbolic path. Various trajectory concepts exist for soft landing a vehicle on the lunar surface; of principal interest are the direct vertical, direct nonvertical and intermediate orbit types. Because of improved site selection and abort capability, the intermediate orbit landing trajectory is far more flexible than a direct landing for either a manned or unmanned soft-lunar landing mission. As a result, this landing mode was utilized in the major portion of the analysis presented in this document.

Because of greater ideal velocity requirements, the direct vertical landing has a lower payload capability than the direct nonvertical (e.g., a gravity turn propulsion descent) landing or the intermediate orbital landing. A more serious disadvantage of the vertical trajectory is the fact that, should the propulsion system fail to ignite at the prescribed time, a collision with the lunar surface is inevitable; this maneuver was therefore disqualified from further consideration for manned missions.

Both the orbital and direct nonvertical maneuvers may be planned so that failure of the propulsion system to ignite does not result in lunar impact but instead returns the vehicle to Earth along a circumlunar trajectory. The choice of landing sites is restricted for the direct landing while the orbital landing allows touchdown at any point on the lunar surface below the parking orbit. Two further advantages of the orbital approach are that it uses techniques developed by assumed previous nonlanding flights, and that it allows reconnaisance of the landing site. A disadvantage of the selected intermediate orbit trajectory is that it requires two additional propulsion system restarts.

Several techniques for landing from circular lunar orbit were investigated. These were categorized as continuous-powered or intermediate-coast phase. For the continuous-powered technique, retrothrust is initiated in the intermediate lunar orbit and continues until the vehicle reaches zero velocity at the lunar surface. The thrust and thrust attitude during descent must be compatible with the orbit height, or the constraints that altitude and velocity reach zero simultaneously cannot be satisfied by a constant-thrust propulsion system. The Intermediate Coast Phase trajectory is characterized by two propulsive applications separated by a coast interval. For optimum execution of this type of descent, a short propulsion phase (small velocity increment) is used to transform the initial circular orbit to a low-periapsis ellipse. The coast phase follows until the vehicle has descended to the trajectory periapsis (i.e., 180 degrees coast). The propulsion system is then reignited and reduces the velocity to zero at the lunar surface. The intermediate-coast type was selected for subsequent studies because it permitted the use of a wide range of F/W ratios and orbit altitudes; this flexibility was not available for continuous-powered descents from the circular lunar orbit. The principal disadvantage of intermediate coast-phase trajectories is the need for an additional engine start.

For the intermediate coast trajectories, use of thrust-to-(Earth) weight ratios greater than 1.0 causes little decrease in ideal velocity requirement. For thrust-to-(Earth) weight ratios below 0.4, ideal velocity requirements increase rapidly as thrust-to-weight ratio is decreased. The velocity requirements are lower for a low specific impulse system because, for a given initial thrust-to-weight ratio (F/W), the average F/W during the landing is higher, caused by more rapid propellant consumption.

Several methods of thrust application were considered for the major deceleration phase of the orbital descent maneuver. The most efficient method, and therefore the method utilized in subsequent analysis, was the thrust-opposing-and-parallel-to-velocity technique.

In a thrust-opposing-and-parallel-to-velocity descent, the F/W ratio and the ellipse pericynthion altitude are related to the landing trajectory shape. The pericynthion altitude must be increased as F/W is reduced; this is caused by the longer powered flight time required to reduce the vehicle energy at low thrust levels. Lunar topography limits the pericynthion altitude to values greater than approximately 30,000 feet, corresponding to a F/W (Earth) of 0.65 or less at the beginning of the descent-from-pericynthion phase.

Analysis of the velocity requirements for takeoff maneuvers exhibited a trend similar to the landing maneuvers. Direct and intermediate-orbit type trajectories were analyzed to determine velocity requirements for each of these techniques. A comparison of the velocity requirements for the two types of Earth-return maneuvers (direct and intermediate-orbit) are presented in Figure 14 . The indirect trajectory requires a slightly greater velocity increment; selection of a lower parking orbit altitude would, however, reduce the indirect mission velocity requirements to values closer to the direct mission values.

LUNAR LANDING AND TAKEOFF PROPULSION REQUIREMENTS

The two primary modes of performing a lunar landing, the direct and lunar-orbit rendezvous (LOR) methods, were analyzed to evaluate their propulsion requirements and to determine the optimum propulsion system characteristics associated with each of these mission modes. The rendezvous mission technique offers the advantage of greater efficiency (i.e., more payload per unit weight of the transfer vehicle), but this advantage is realized only if the combination of landing site and stay-time is such that significant plane-changes by the ascent vehicle and/or the parent vehicle are avoided. The analyses of propulsion requirements have in part been based on vehicles of the Apollo size or Saturn C-5 capability; the parametric data and results presented are, however, applicable to larger, later-generation vehicles.

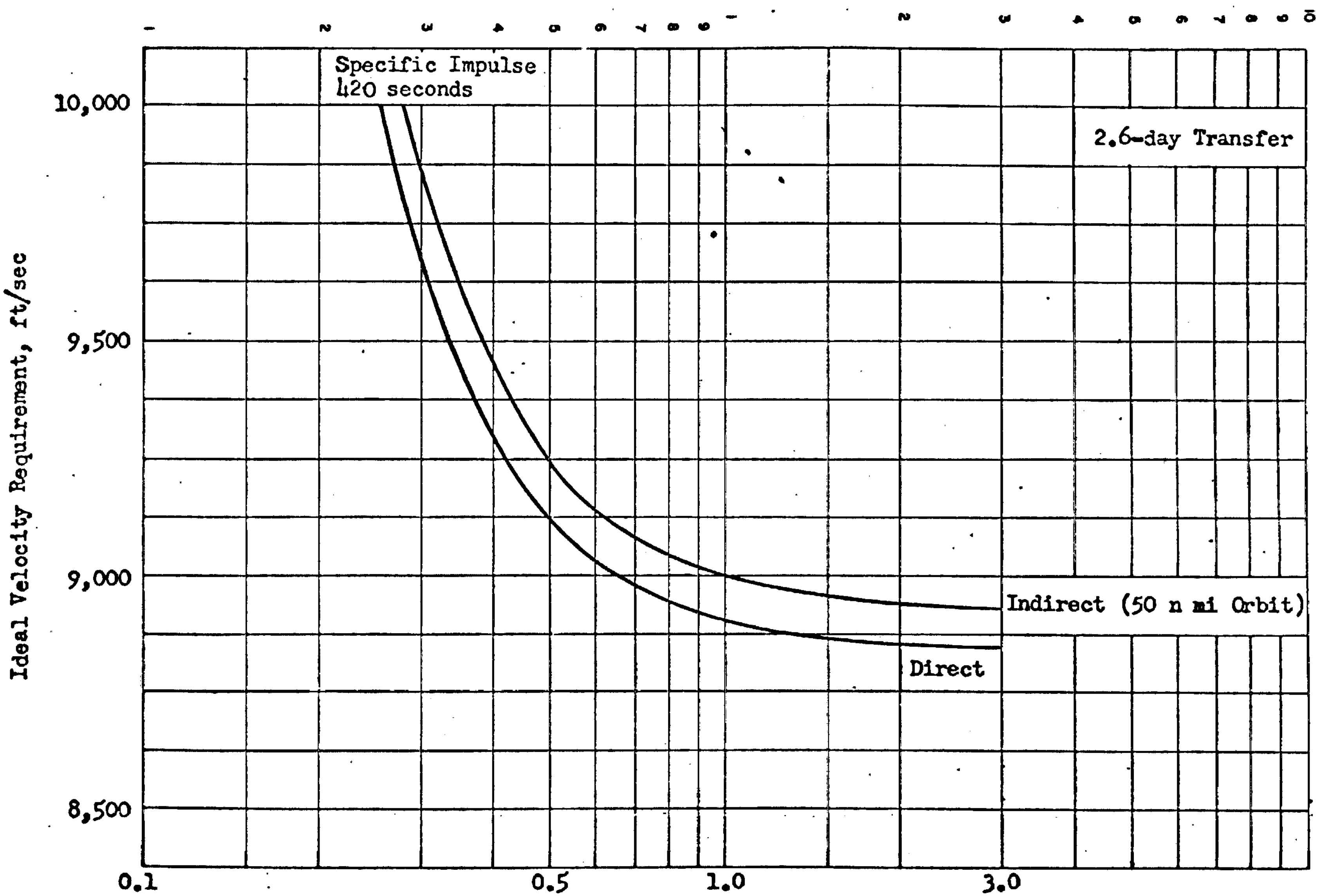


Figure 14. Ideal Velocity Requirement for Lunar Takeoff to Moon-Earth Coast Trajectory

The total velocity requirement of a direct landing system is the sum of velocity additions (chronologically) for midcourse correction ($\sim$ 150 ft/sec), circular orbit establishment ($\sim$3200 ft/sec), orbit eccentricity change ($\sim$ 60 ft/sec), velocity cancellation ($\sim$ 5700 ft/sec) and hovering/translation (from 200 to 1000 ft/sec). In addition, a propellant reserve equivalent to approximately 300 ft/sec ($\sim$ 3 percent) is included. Thus, the overall velocity requirement is between 9500 ft/sec and 10,500 ft/sec.

The selection of thrust level for lunar landing is governed principally by the exchange between velocity requirements and propellant-dependent weights (each of which decreases as thrust-to-weight ratio increases) and engine and thrust structure weights (which decrease as thrust-to-weight ratio decreases). The velocity requirements for intermediate orbit landing are very similar to the direct nonvertical landing. The intermediate orbit landing trajectory is more flexible and therefore was used in the analysis. For thrust level selection, the thrust-dependent weight factor is of primary importance, as shown in Figure 15 . Optimum thrust-to-weight ratio decreases from 0.475 when the thrust-dependent factor is 0.02 lb/lb thrust to 0.3 when the thrust-dependent weight factor is 0.06. A wide range of thrust-dependent weights must be considered since redundant systems may be employed, and the degree of engine redundancy strongly affects engine weight factor.

The fixed weight, tank weight, hovering ΔV, transfer time, interstage weight, and specific impulse are all factors which do not in general strongly influence thrust level selection. Also, the penalty for operation at an off-optimum thrust level is not severe. For example, for a typical O_2/H_2 system, vehicle gross weights within 1 percent of the minimum (which occurs in this instance at a thrust-to-weight ratio of 0.34) can be achieved with thrust-to-weight ratios from 0.22 to 0.58.

The thrust-to-weight ratio of the vehicle at the end of the main descent maneuver represents the initial condition for translation, hovering and final descent maneuvers. The variation of terminal thrust-to-weight ratio for a direct landing maneuver is shown in Figure 16 . To achieve a 1:1 vehicle thrust-to-weight ratio (necessary for constant altitude hovering), an engine throttling ratio equal to the terminal thrust-to-lunar weight ratio is required. For satisfactory control during the terminal landing phase, it may be necessary to throttle the landing engine to thrust-to-weight ratios substantially below 1:1, and engine designs must include an allowance for this consideration.

For the direct landing, based on the vehicle and trajectory characteristics considered, a thrust-to-weight ratio of approximately 0.45 is desirable. An engine throttling capability of 10:1 would provide sufficient thrust control for performance of hovering and translation maneuvers near the lunar surface.

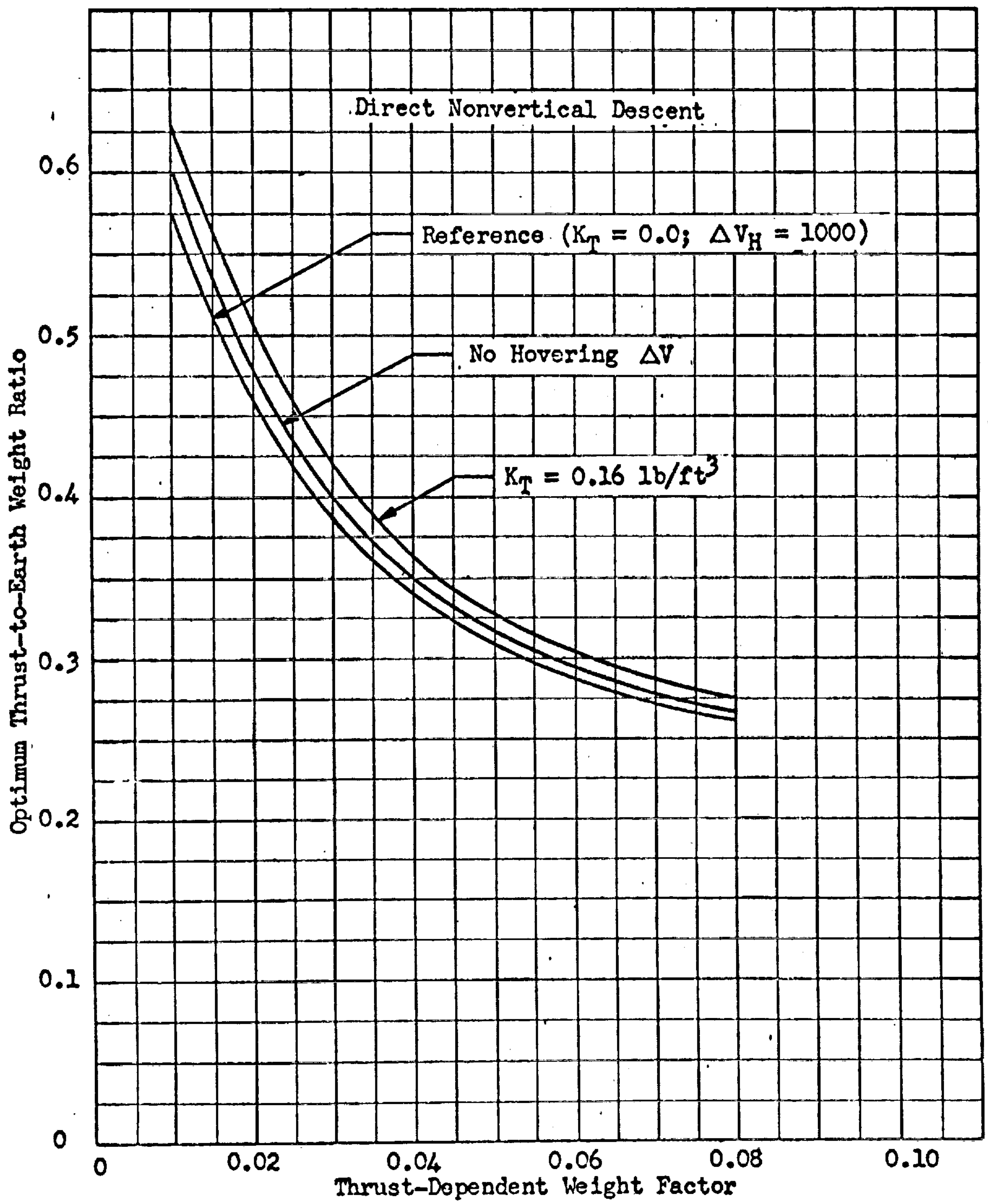


Figure 15. Thrust Selection for Lunar Landing

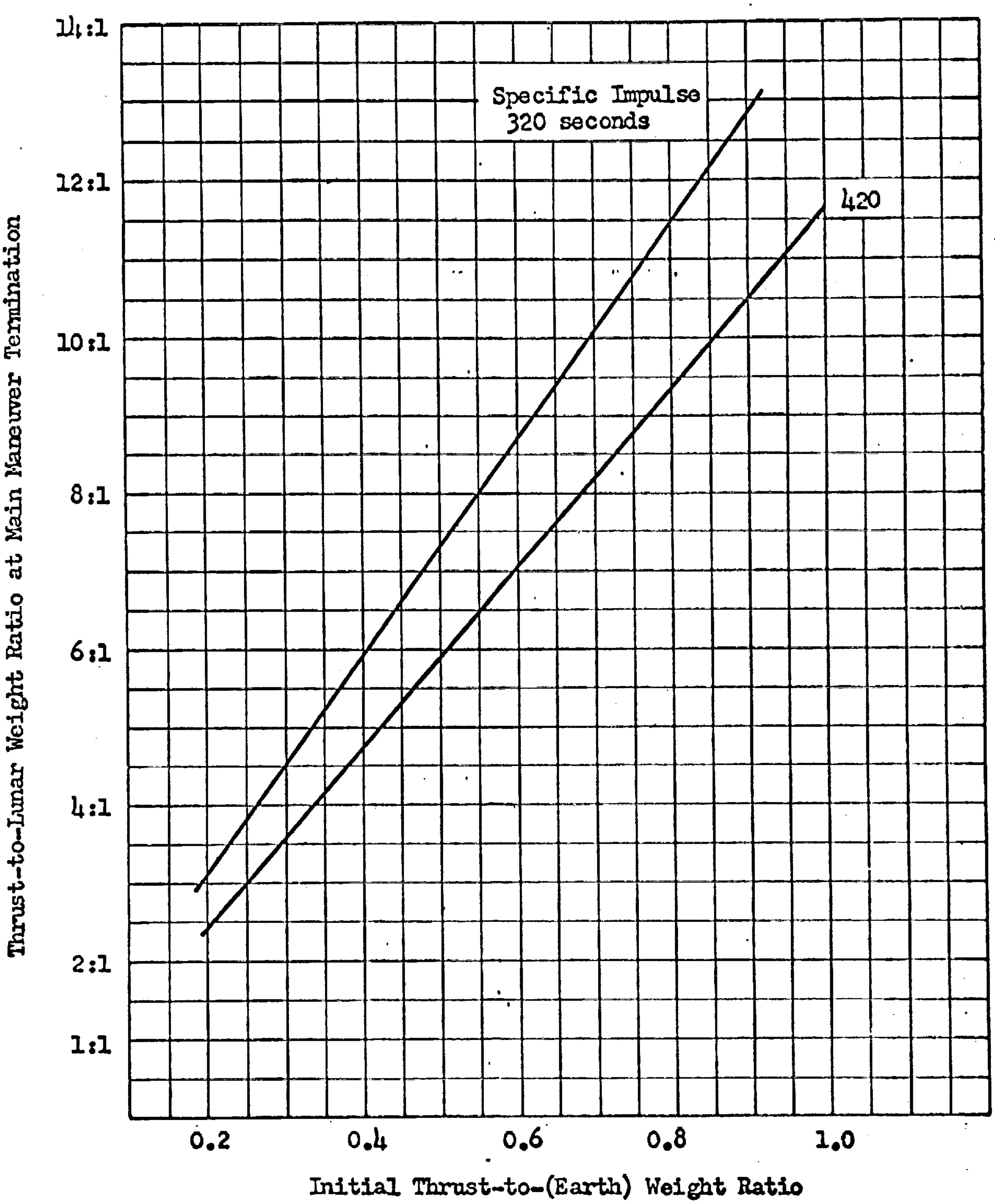


Figure 16. Throttling Ratio Required to Achieve Thrust Equal to Lunar Weight: Direct Lunar Landing

To illustrate how the selection of propulsion systems governs the feasibility of a lunar mission within the restrictions imposed by the launch vehicle capability and the Earth-return payload requirement, Figure 17 indicates landing stage capabilities and takeoff stage requirements for a direct lunar landing mission. As indicated, the mission can be satisfactorily accomplished, (i.e., the landed payload weight exceeds the required takeoff weight) by the use of a pump-fed O_2/H_2 landing engine or by the use of pressure fed O_2/H_2 systems for both landing and takeoff.

In the Lunar orbit rendezvous method, the landing propulsion system (single stage or multistage) must accomplish the descent, translation, takeoff, and rendezvous maneuvers. Various one and two stage configurations were considered for the landing-from-orbit/return-to-orbit function. For a 35,000-pound gross weight landing vehicle detached from the orbiting parent vehicle, useful payloads ranging from approximately 4000 pounds (for a single stage, noncryogenic propellant, pressure-fed system) to approximately 10,000 pounds (for a two-stage, O_2/H_2, pump-fed system) were obtained. The nominal vehicle, a two-stage, noncryogenic propellant, pressure-fed system, delivered a payload slightly in excess of 6000 pounds to the lunar surface and back to orbit.

Parametric design studies and thrust optimization studies of various LOR systems were made. An example of the results are shown in Table 24. The throttling ratio shown is to achieve a 1:1 thrust-to-lunar weight for hovering.

Optimum thrust-to-weight ratio (F/W) for the propulsive maneuvers employed in the two landings presented in Table 25.

TABLE 25

THRUST SELECTION

Maneuver	Optimum Thrust-to-Earth Weight Ratio: Cryogenic System	Optimum Thrust-to-Earth Weight Ratio: Noncryogenic System
Direct Landing	0.45	--
Landing-from-Orbit	0.55	0.60
Takeoff-to-Orbit	0.65	0.75
Combined Landing-from-Orbit and Takeoff-to-Orbit	0.55	0.65

SATURN C-5 LUNAR LOGISTICS VEHICLE

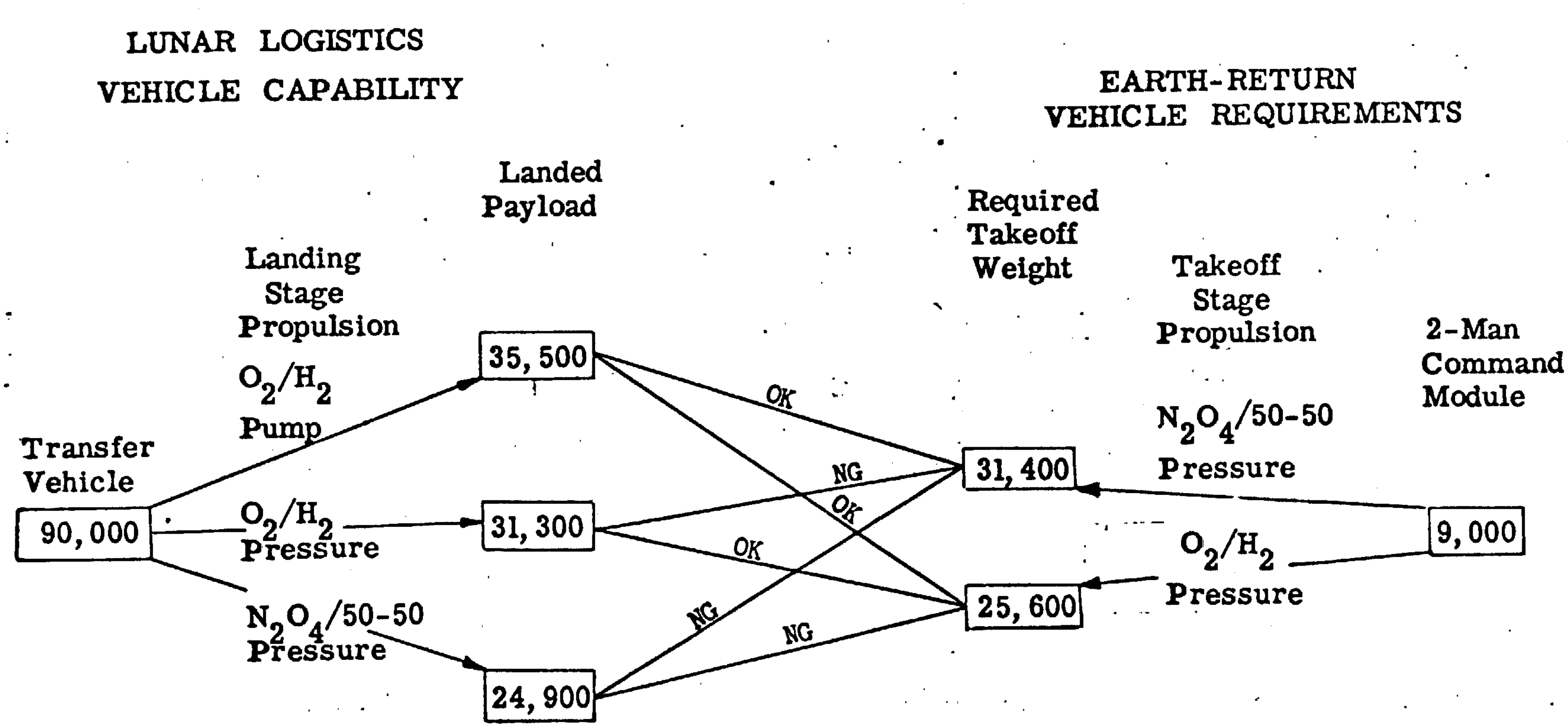


Figure 17

TABLE 24

O_2/H_2 LUNAR LANDING/TAKEOFF VEHICLES

	Two Stage		Single Stage
	Landing	Takeoff	
Initial Weight, pounds	35,000	18,600 pump-fed* 17,300 pressure-fed	35,000
Specific Impulse, seconds	420	420	420
Thrust-Dependent Weight Factor	0.025 0.040	0.025 0.040	0.025 0.040
Propellant-Dependent Weight Factor	0.14 0.21	0.14 0.21	0.14 0.21
Optimum Thrust, pounds	19,300 17,900	11,700 9,700	19,100 17,200
Payload, pounds	18,600 17,300	9,900 8,500	8,600 6,800
Thrust for Payload Within 1 percent of Maximum, thousand pounds	15.0 - 26.6 14.4 - 22.4	9.1 - 17.3 8.0 - 13.1	15.6 - 24.1 14.9 - 20.1
Throttling Ratio for Optimum Thrust System, pounds	5.0:1 4.6:1		5.0:1 4.5:1
Propellant Weight in Optimum Thrust System, pounds	13,670 13,700	7,970 7,480	13,680 + 8660** 13,730 + 8660
Duration of Optimum Thrust System, seconds***	373 397	286 324	377 + 190 409 + 212

* Pairs of number indicate pump/pressure throughout Table
** Division separates landing and takeoff phases
*** Includes 94 seconds of hovering for landing phases

The effect of thrust-to-weight ratio on payload was not pronounced; a wide range of F/W values can be used without significantly penalizing vehicle payload capability.

ERROR ANALYSIS FOR LUNAR LANDING-FROM-ORBIT MANEUVER

A study was performed to evaluate the terminal position errors that are experienced when a propulsive landing maneuver from lunar orbit is not executed precisely in accordance with nominal conditions. The errors considered were deviations in thrust, early or late initiation of the landing maneuver, and angular displacement between the nominally-parallel thrust and velocity vectors.

The nominal conditions employed in this analysis are tabulated below:

Initial Thrust-to-	
Earth Weight	0.4
Local Weight	2.4
Burnout Thrust-to-	
Earth Weight	0.7
Local Weight	4.4
Specific Impulse, seconds	315
Periapsis Altitude, feet	71,000
Periapsis Velocity, ft/sec	5,704
Ideal Velocity Increment, ft/sec	5,984

The specific impulse value reflects the use of noncryogenic propellants to satisfy the velocity requirements of the mission.

For the range of errors considered, the ideal velocity requirement to decelerate the landing vehicle to rest was essentially unaffected by deviations from nominal conditions, amounting to only ± 20 ft/sec with respect to the nominal case. The hover position, however, was sensitive to small variations in thrust, ignition-time and vector alignment. Representative values of final position errors, with respect to the nominal final position, are presented in Table 26 . These results do not include a small effect associated with lunar rotation. The indicated propulsion and trajectory errors are typical of expected deviations from nominal conditions; the altitude errors suggest the selection of a nominal hover point 5000 feet above the lunar surface.

TABLE 26

HOVER POINT POSITION ERRORS

	Δ Altitude, feet	Δ Range, n mi
+2 percent Thrust, I_s Constant	+2710	-3.48
+2 percent Thrust, +2 percent I_s	+1890	-2.88
Ignition 20 seconds early	-1650	-18.5
+0.5 degrees Misalignment	-4440	+0.10

It is significant to note that the translation and descent studies described below indicate that the propulsion requirements to perform a 1- n mi translation near the lunar surface is approximately 800 ft/sec and to accomplish a 1000-foot vertical descent to the lunar surface is on the order of 200 ft/sec. These penalties, considered in conjunction with the Table 26 data, strongly suggest that corrective measures such as engine throttling be employed during the landing maneuver to obtain direct transit to the desired landing site.

The fact that misalignment has little effect on range deviation suggests that deliberate misalignment does not offer an efficient means of correcting range errors introduced by other factors. Extreme values of misalignment might provide substantial range corrections, but only at the expense of large penalties to ideal ΔV and required hover altitude.

MISSION ABORT

A study was conducted to evaluate abort propulsion requirements during the main propulsive phase of a landing-from-lunar orbit. The configurations analyzed were single stage vehicle with sufficient capability for descent-from-orbit and launch-to-orbit and a two-stage vehicle which assigned the two maneuvers to two distinct propulsion systems.

The single stage vehicle provided adequate propulsive capability (both thrust and velocity capability) to return the vehicle to its initial 50-n mi orbit from any point along the landing trajectory. Altitude loss during the abort maneuver was minimized by directing the thrust vector vertically upward, although the ΔV requirement was thereby maximized, while the ΔV requirement was minimized by permitting the abort trajectory to graze the lunar surface. For example, an abort initiated at 25,000 feet descended no lower than 20,600 feet by orienting the thrust vector vertically upward; the corresponding ΔV was 6200 ft/sec. By utilizing a trajectory that grazed the lunar surface, the ΔV requirement was 4310 ft/sec.

For a two-stage vehicle, the propulsive margin is even greater than for a single stage vehicle if a reasonable portion of the landing stage is available to assist in the abort maneuver. More significant is the case where only the takeoff stage is available (i.e., noncatastrophic failure of the landing stage). The restrictions imposed on the lunar takeoff stage if it must be able to perform the abort maneuver at any point along a descent trajectory were investigated. Ideal velocity increments and initial thrust-to-weight ratios required by the takeoff stage for the abort maneuver were determined for any point along a typical descent trajectory. These requirements were based upon a minimum energy abort trajectory in which the vehicle, during the early portion of the abort maneuver, descends to a point near the lunar surface, then circularizes while traversing a short distance at constant altitude before ascending to the lunar orbit. During this interval, the vehicle accelerates to sufficient velocity for a coast phase to the 50-n mi orbit.

Based upon this type of minimum energy trajectory, the abort stage, or lunar takeoff stage, requires an initial thrust-to-weight ratio equal to or greater than the maximum vertical thrust-to-weight component of the landing stage. This conclusion applies to a descent trajectory having zero hover altitude. If a positive-hover altitude is included for final descent and translation to the lunar surface, the thrust-to-weight for the abort stage (i.e., takeoff stage) could be decreased, thereby permitting greater latitude in selecting a takeoff thrust-to-weight ratio.

To fulfill the design requirements for a takeoff stage which performs the abort maneuver, a typical optimized vehicle landing trajectory was assumed. Based on these assumptions and zero-hover altitude, the takeoff stage was found to require an initial thrust-to-(Earth) weight ratio of 0.63 and a velocity requirement of 5725 ft/sec. If allowances are made for hover altitudes of 100 feet or 1000 feet, then the takeoff-stage initial thrust-to-weight requirements are reduced to 0.49 (near optimum for maximum performance) and 0.37 respectively. Simultaneously, the ideal velocity requirements for the takeoff stage which performs the abort maneuver with these lower F/W ratios increase to approximately 5835 and 6100 ft/sec.

NEAR-SURFACE TRANSLATION

For a nonaerodynamic planetary landing mission, it may be desirable, following the major deceleration maneuver, to perform a translation maneuver prior to the actual landing. Some of the reasons for this requirement include refined landing-site selection, planetary surface survey and the need to reach a specific point on a planet surface. Several methods for applying rocket propulsion systems to the performance of this maneuver are possible. The two basic propulsion methods are defined as ballistic or continuous, with the latter capable of providing horizontal translation.

In the study presented, methods of performing the translation maneuver were surveyed, and a detailed analysis of continuous-powered, single-engine systems was conducted.

Review of the analysis and results indicates that the ballistic system offers the most favorable propellant economy for downrange translation (approximately 25 percent less propellant than a single-engine, continuous-powered system for a given maneuver). However, several disadvantages exist: engine restarts are required; large vehicle tilt angles can exist; the downrange distance cannot be changed enroute; and high altitude trajectories preventing surveillance can result.

For the multiengine horizontal translation system, no tilting of the vehicle is required, and the single main engine thrust can be maintained at a near constant level. (Throttling is only necessary to compensate for propellant consumed.) However, the system has the disadvantage of requiring additional restartable engines, and the auxiliary engine must be located at the vehicle cg to prevent vehicle rotation, or the main engine must be gimbaled.

The single engine, continuous-powered translation method appears desirable with respect to simplicity, reliability, and versatility. This method eliminates the requirement of engine restart. Use of a throttleable main engine allows a continuous constant altitude, but requires thrust adjustment during the maneuver. The optimum angles for single-engine translation maneuvers (45 degrees, if no intermediate coast phase is employed; 30 degrees, with coast) are somewhat high for short translation distances, and for long translation distances, the horizontal velocity with these tilt angles might be excessive for ground surveillance. The propellant-consumption decrease obtainable by the use of a coast phase does not appear to warrant the additional rotation maneuvers required.

The investigation of constant thrust translation showed that translation with either increasing, decreasing, or approximately constant altitude can be achieved with a constant engine thrust. However, the thrust at initiation of the maneuver must be the amount specified to achieve the desired translation trajectory. An intermediate horizontal coast phase between the acceleration and deceleration phases was examined and found to require throttling to prevent altitude change, and, in general, did not offer significant benefits.

The engine gimbaling conditions (angles and rates) do not appear to be a critical factor. Changes in engine gimbaling produce only very slight changes in the overall translation maneuver. Vehicle orientation (tilt) angles are not critical for short translation distances, but in general have a pronounced effect on translation trajectory characteristics.

For a representative, constant-altitude translation maneuver of 3000 feeet, a vehicle employing the single-engine, continuous-powered technique without a coast phase requires 400 ft/sec of ideal velocity capability.

LUNAR LANDING FINAL DESCENT PHASE

The propulsion requirements for the major braking from Earth-transfer or lunar orbit, translation to desired touchdown point, and vertical descent-to-surface maneuvers for a lunar landing are widely different with respect to both velocity increment and thrust level. However, it is quite likely that a single propulsion system will be employed to perform all three maneuvers; therefore, the selected propulsion system characteristics must be adequate to satisfy the individual requirements of each maneuver.

The vertical descent phase following the translation maneuver requires minimum propellant expenditure when it is performed as a two-thrust-level operation, initially utilizing the lowest (maximum throttle) and then the highest (zero throttle) thrust levels within the propulsion system capability. The effects of possible variations in these independent variables on velocity requirements are sizable (up to 100 percent ΔV variation) when considered solely in the context of the descent maneuver, but small as a fraction of the overall stage propulsion capability.

The ideal velocity capability required for performance of a vertical descent maneuver to the lunar surface

1. Is approximately 75 ft/sec for a typical case in which initial altitude is 200 feet, initial descent rate is zero, maximum thrust-to-weight ratio is 6 (representative of the burnout thrust-to-lunar weight ratio of an optimized landing-from-orbit or direct landing stage) and throttling ratio is 10:1

2. Decreases as throttling ratio increases, though for most cases, throttling capability beyond 10:1 provides only small benefits

3. Increases with increasing initial altitude (e.g., 120 ft/sec, if the vehicle described above initiates descent from 500 feet instead of 200 feet)

4. Is less for relatively low nonzero initial descent rates than it is for a zero initial rate of descent. An optimum initial descent rate exists, and is dependent on initial altitude, maximum thrust and throttling ratio

5. Is a function of maximum F/W, and displays an optimum which is dependent primarily on throttling ratio

The maximum velocity achieved during a vertical descent maneuver increases with increasing throttling ratio. However, deliberate reduction of maximum velocity, attained by employing less-than-available throttling, imposes a propellant penalty on the vehicle system.

TOUCHDOWN STABILITY

An analysis was conducted to evaluate the trajectory, vehicle and terrain factors governing touchdown stability of an assumed lunar landing vehicle. The vehicle stability criterion was based on the condition that the angular kinetic energy of the vehicle at impact be sufficient to rotate the vehicle to an unstable position.

The vehicle impacts the surface with an initial kinetic energy which is the result of a residual vehicle velocity (V). Since the landing legs have the ability (by design) to absorb energy, the energy associated with the velocity component (V_L) along the leg is assumed to be completely absorbed. The energy acting to tip the vehicle is associated with the velocity component (V_R) perpendicular to the leg. This energy is equated to the potential energy required to lift the center of gravity (cg) to the point of instability; that is, the vehicle rotates about the point of impact until the cg swings through the vertical (point of instability), and the vehicle falls on its side.

The results obtained define combinations of vertical and horizontal velocity components which permit stable touchdown. As indicated on the horizontal vs vertical velocity component grid shown in Figure 18, representing the case of foreleg impact, the region of stable impact is dependent on the angle of impact and the vehicle moment of inertia about the impact point. Stability is also governed by the inclination of the landing surface (significant only for hindleg impact situations), the height of the vehicle cg and the distance from the impact point to the vehicle longitudinal axis.

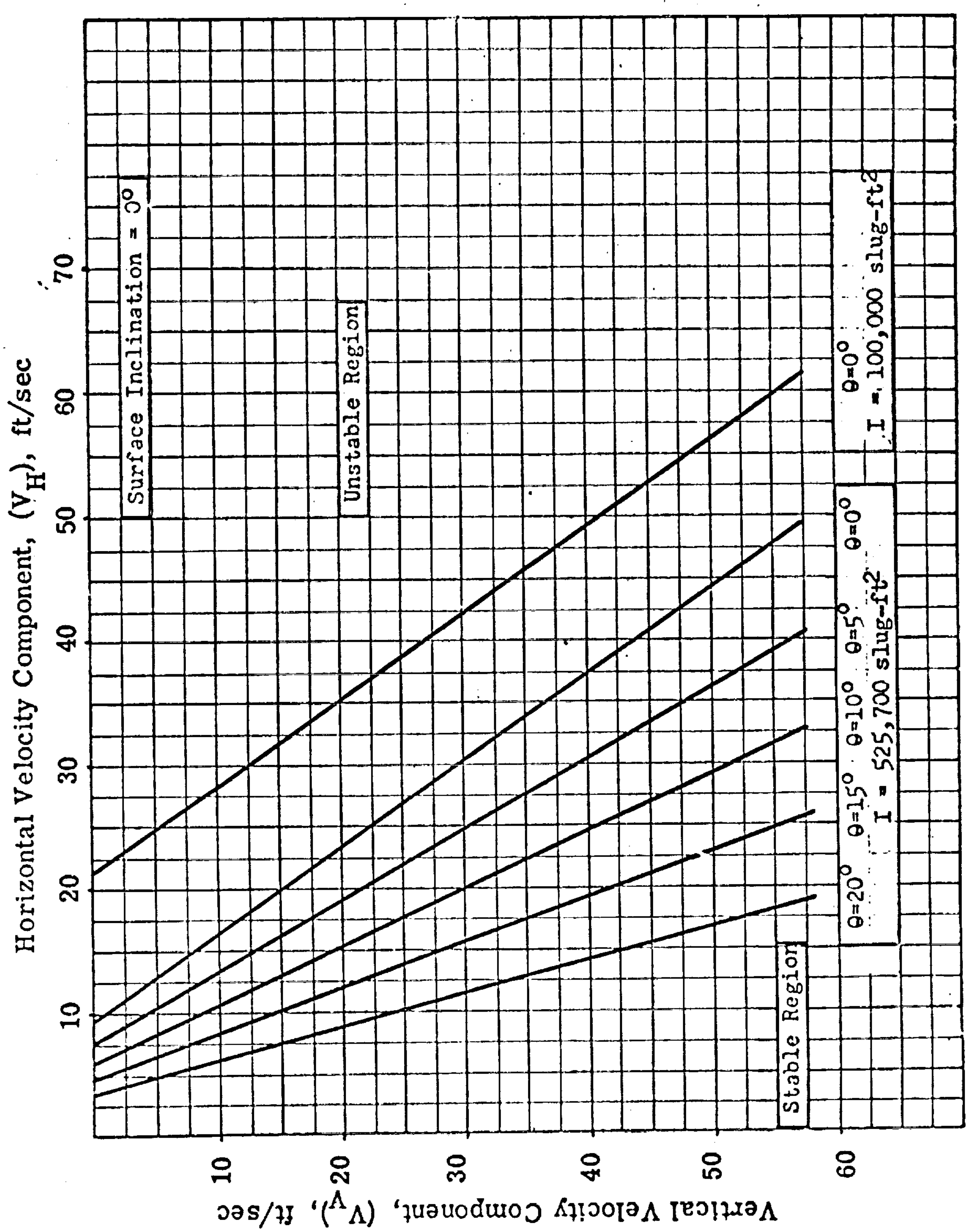


Figure 18. Fore-leg Impact of Lunar Landing Vehicle

EARTH-MERCURY MISSIONS

MERCURY TRANSFER PHASE

Earth-Mercury trajectories were computed to determine an optimum launch date and trip time for a soft landing mission. The objective of the study was to minimize the sum of the propulsion requirements for the Earth-departure and Mercury-arrival phases of the mission, and to evaluate the penalties incurred by launching close to, but not precisely on, the optimum launch date.

The selected trajectory is a 90-day transfer launched on 10 May 1973. The Earth-departure hyperbolic excess velocity is 31,000 ft/sec, corresponding to a 22,000 ft/sec impulsive velocity increment from a 300-n mi, circular Earth orbit. The hyperbolic arrival velocity at Mercury is 27,000 ft/sec; a 21,000 ft/sec impulsive velocity change decelerates the vehicle into a 300-n mi circular Mercurian orbit.

The hyperbolic arrival velocity is less than 30,000 ft/sec during the interval from 6 May 1973 to 13 May 1973. Cyclic repetition of this optimum trip is impaired by the eccentricity of the Mercurian orbit about the Sun (0.206) and the inclination of the Mercurian orbit to the ecliptic (7 degrees); a fairly similar optimum transfer cannot be achieved until 1986. In the intervening period, minimum hyperbolic arrival velocities are on the order of 50,000 ft/sec.

MERCURY ORBIT ESTABLISHMENT

Because the 23,560 ft/sec impulsive ideal velocity requirement for establishment of a 300-n mi Mercury orbit from a 30,000 ft/sec hyperbolic arrival velocity is a rather high ideal velocity to be supplied by a single-stage using conventional chemical propellants, a study was conducted to evaluate the payload advantage of a vehicle with tank-staging or a two-stage vehicle in comparison to a reference single-stage vehicle. The purpose of the study was also to determine the optimum thrust-to-weight ratios for the three systems.

The results of this study indicate that the single-stage orbit establishment ideal velocity requirement is not significantly different from that for a vehicle which stages tanks. The ideal velocity requirement for a two-stage vehicle, however, can vary significantly from that for a one-stage vehicle with the same initial thrust-to-weight; the magnitude of this difference depends upon the thrust-to-weight ratio of the second stage.

The data presented in Table 27 indicate that a single stage or a tank-staging orbit-establishment vehicle arriving with a 30,000-ft/sec hyperbolic velocity will have an optimum thrust-to-weight ratio of about 0.3. The optimum thrust-to-weight ratio is about 0.5 for both stages of a two-stage vehicle. By the results summarized in Table 28 it is shown that the payload-to-gross-weight ratio of a two-stage vehicle is 41 percent higher than that of a single stage vehicle. For a tank-staging vehicle (tanks jettisoned four times) the payload-to-gross weight ratio is 48 percent higher.

The three types of vehicles were compared at various hyperbolic velocities. As shown in Table 29 , the higher the hyperbolic velocity the more advantage two-stage and tank-staging vehicles have over a single-stage vehicle.

MERCURY ORBITAL LANDING AND TAKEOFF

The absence of an atmosphere about the planet Mercury dictates that landing maneuvers be performed entirely propulsively; there is no recourse to aerodynamic assistance. The propulsion requirements for landing from orbit by means of single stage vehicles were obtained by computation of simulated landing trajectories, and the results were applied to an investigation to determine the optimum vehicle thrust-to-weight ratio for the maneuver.

The computed velocity data were utilized in conjunction with representative vehicle characteristics to determine the optimum thrust-to-weight ratios for typical non cryogenic and cryogenic propellant landing vehicles. The results indicate selection of a thrust-to-Earth weight ratio between 0.8 and 0.9, and a propulsion requirement of approximately 11,400 ft/sec ideal velocity increment for deceleration to zero velocity at a point near the surface of Mercury.

Simulated takeoff trajectories were computed to determine ideal velocity requirements for the takeoff maneuver as a function of thrust-to-weight ratio and specific impulse. A thrust-to-weight ratio optimization was not performed, but the steeper increase in ΔV as F/W decreases (as compared to the landing data) implies that the optimum F/W is greater than the value for a landing system.

The velocity data obtained represent the basic requirements for performance of the major propulsive phases of landing and takeoff maneuvers. As indicated previously for lunar landing and takeoff vehicles, stage capability must be sufficient not only for these phases, but for performance of secondary propulsive maneuvers as well. Based on studies of lunar near-surface translation and vertical descent, and considering the difference

TABLE 27
MERCURY ORBIT ESTABLISHMENT VEHICLE
OPTIMUM THRUST-TO-EARTH WEIGHT RATIO

Mission: Orbit-Establishment from 30,000 ft/sec Hyperbolic Arrival Velocity

Vehicle	Optimum Initial F/W*	Initial F/W Range for-2 Percent Payload
Single Stage	0.31	0.17 → 0.53
Two-Stage		
$F/W_2 = 0.5\ F/W_1$		0.66 → 1.20
$F/W_2 = F/W_1$	0.5	0.31 → 0.82
$F/W_2 = 2\ F/W_1$		0.18 → 0.40
$F/W_2 = 4\ F/W_1$		0.15 → 0.20
∞ Tank Staging	0.24	0.12 → 0.49

TABLE 28
MERCURY ORBIT ESTABLISHMENT VEHICLE
IDEAL VELOCITY INCREMENT AND PAYLOAD

Mission: Velocity Orbit-Establishment from 30,000 ft/sec Hyperbolic Arrival

Vehicle	Approximate Ideal Velocity Requirement (ft/sec)	Payload to Gross Weight Ratio	Percent of Single Stage Payload
Single Stage ($F/W = 0.3$)	24,000	0.081	100
Two-Stage ($F/W_1 = 0.5$) ($F/W_2 = 0.5$)	24,000	0.114	141
Single Stage Tanks Jettisoned One Time ($F/W = 0.3$)	24,000	0.107	132
Single Stage Tanks Jettisoned Four Times ($F/W = 0.3$)	24,000	0.120	148

TABLE 29
EFFECT OF HYPERBOLIC ARRIVAL VELOCITY ON PAYLOAD

Vehicle	Payload-to-Gross-Weight Ratio				Percent of V_H = 30,000 Payload			
V_H**	20,000	30,000	40,000	50,000	20,000	30,000	40,000	50,000
Single Stage	0.251	0.081	-0.006	-0.048	310	100	0	0
Two-Stage	0.265	0.114	0.041	0.012	230	100	36	10
∞ Tank Staging	0.279	0.128	0.055	0.021	220	100	43	16

* Thrust-to-Weight Ratio
** V_H = Hyperbolic Arrival Velocity, ft/sec

in gravity constants, allowances of 600 ft/sec for translation and 300 ft/sec for descent are indicated. Efficient performance of the descent maneuver also requires that the landing stage propulsion system be capable of approximately 9:1 throttling.

ERROR ANALYSIS FOR MERCURY LANDING-FROM-ORBIT MANEUVERS.

An investigation was conducted to evaluate the terminal position errors that are experienced when a propulsive landing maneuver from Mercury orbit is not executed precisely in accordance with nominal conditions. The errors considered were deviations in thrust, early or late initiation of the landing maneuver, and angular displacement between the nominally-parallel thrust and velocity vectors.

The nominal conditions employed in the analysis are tabulated below:

Initial Thrust-to-	
Earth Weight Ratio	0.9
Local Weight Ratio	2.4
Burnout Thrust-to-	
Earth Weight Ratio	2.0
Local Weight Ratio	5.3
Specific Impulse, seconds	420
Periapsis Altitude, feet	120,000
Periapsis Velocity, ft/sec	10,400
Ideal Velocity Increment, ft/sec	10,790

The specific impulse value reflects the use of high-energy cryogenic propellants for the Mercury landing mission.

For the range of errors considered, the ideal velocity requirement to decelerate the landing vehicle to rest was essentially unaffected by deviations from nominal conditions, amounting to only ± 20 ft/sec with respect to the nominal case. The hover position, however, was sensitive to small variations in thrust, ignition-time and vector alignment. Representative values of final position errors, with respect to the nominal final position, are presented in Table 30 . These results do not include a small effect associated with Mercurian rotation. The indicated propulsion and trajectory errors are typical of expected deviations from nominal conditions; the altitude errors suggest the selection of a nominal hover point 6000 feet above the Mercury surface.

TABLE 30

HOVER POINT POSITION ERRORS

	Δ Altitude, feet	ΔRange, n mi
+2 Percent Thrust, I_s Constant	+3130	-4.78
+2 Percent Thrust, +2 Percent I_s	+1780	-3.65
Ignition 20 Seconds Early	-3050	-34.0
+0.5-degree Misalignment	-5760	+0.13

Previous studies of lunar near-surface translation and descent have established that the propulsion requirements to perform a 1-n mi translation near the lunar surface is approximately 800 ft/sec and to accomplish a 1000 foot vertical descent to the lunar surface is on the order of 200 ft/sec. These penalties, considered in conjunction with the Table 30 data, strongly suggest that corrective measures such as engine throttling be employed during the landing maneuver to obtain direct transit to the desired landing site.

The fact that misalignment has little effect on range deviation suggests that deliberate misalignment does not offer an efficient means of correcting range errors introduced by other factors. Extreme values of misalignment might provide substantial range corrections, but only at the expense of large penalties to ideal ΔV and required hover altitude.

ENGINE PARAMETER OPTIMIZATION

A propulsion system parameter study was conducted to determine the optimum designs for propulsion systems applicable to extraterrestrial landing missions. The propulsion parameters considered were chamber pressure (P_c), expansion area ratio (ϵ), and thrust chamber mixture ratio (MR).

A total of 54 basic propulsion models were considered; these were the result of selecting three propellant combinations (O_2/H_2, F_2/H_2, and NTO/50-50), three velocity increments (6,000 ft/sec, 14,000 ft/sec, and 22,000 ft/sec), two thrust-to-weight ratios (0.3 and 0.8), and three thrust levels (5,000 lbs, 50,000 lbs, and 500,000 lbs). These values are representative of requirements determined for various possible landing mission maneuvers.

Several pertinent assumptions regarding system configurations are as follows:

1. Pump-fed systems for possible manned (50,000 and 500,000-pound thrust) vehicles and pressure-fed systems for unmanned (5,000-pound thrust) applications.

2. Regeneratively cooled 80-percent bell nozzles; fully cooled for pump-fed systems and cooled to an expansion area ratio of 20:1, with ablative cooling thereafter, for pressure-fed systems.

3. Single nozzle configurations with specific impulse efficiencies of 0.940 for pressure-fed O_2/H_2 and NTO/50-50, 0.945 for pressure-fed F_2/H_2, 0.950 for pump-fed O_2/H_2 and NTO/50-50 and 0.955 for pump-fed F_2/H_2 systems.

In addition, the interstage structure was designed for structural adequacy rather than as a meteorite shield (which would have made it about two or three times as heavy). This assumption leads to substantially higher values of optimum expansion area ratio than are obtained by the alternative assumption.

Optimum values of P_c, ϵ and MR were determined for each model; additionally, ϵ was fixed at a value of 50:1 in each case, and P_c and MR were optimized. The optimum parameters for selected pump-fed and pressure-fed systems are presented in Table 31 . The permissible range over which each parameter can be varied without causing a payload loss in excess of one-half and one percent are also shown. The data emphasize the general conclusion that propulsion system operating parameters can be widely varied without imposing a significant payload penalty on a vehicle.

TABLE 31

EFFECT OF OFF-OPTIMUM DESIGN

System Type	Operating Parameter	Allowable Parameter Increment / Percent Payload Loss 0.5	1.0	Optimum Value
O_2/H_2 Pressure-Fed	P_c, psia	+25	+35	55
	ϵ	-80	-100	200
F = 5000 pounds	MR	$\pm$0.9	$\pm$1.2	6.20
F_2/H_2 Pressure-Fed	P_c	$\pm$22	$\pm$35	80
	ϵ	-60	-90	200
F = 5000 pounds	MR	$\pm$2.3	-3.8	16.90
NTO/50-50 Pressure-Fed	P_c	+40	+60	130
	ϵ	-110	-145	310
F = 5000 pounds	MR	$\pm$0.13	$\pm$0.20	2.18
O_2/H_2 Pump-Fed	P_c	-450	-550	1360
	ϵ	-200	-275	450
F = 50,000 pounds	MR	$\pm$1.0	$\pm$1.4	6.90
F_2/H_2 Pump-Fed	P_c	-650	-900	1770
	ϵ	-230	-270	410
F = 50,000 pounds	MR	-3.2	-4.6	17.35
NTO/50-50 Pump-Fed	P_c	-600	-800	1800
	ϵ	-175	-230	471
F = 50,000 pounds	MR	$\pm$0.12	$\pm$0.18	2.21

Mission ΔV = 14,000 ft/sec, F/W = 0.3, unrestricted area ratio

APPENDICES

APPENDIX A: EXTRATERRESTRIAL ENVIRONMENTS

A review of available data describing the environments of the Earth, moon, Mars, Venus and Mercury was conducted to provide the information required for evaluation of environmental effects on lunar and planetary landings. These data are summarized in Table 32. Of primary importance is the presence of an atmosphere about the planets, Earth, Mars and Venus in contrast to the vacuum surrounding the moon and Mercury.

An investigation of the interactions between atmospheres and rocket exhausts and between surfaces and rocket exhausts indicated 1) that the inertness (predominantly N_2 and CO_2) of the Martian and Venusian atmospheres precludes chemical reaction with rocket exhaust products and 2) that for a sufficiently soft surface, as may exist at the moon, a sizable crater can be formed by the exhaust jet of a descending vehicle. Impairment of visibility by the exhaust plume was determined to be insignificant for a lunar landing.

APPENDIX B: VEHICLE REQUIREMENTS FOR INTERPLANETARY MISSIONS

In the future, vehicles will be designed to land instrumented probes on the planetary surfaces and subsequently return them to the Earth. Though this type of mission is sufficiently far in the future that one cannot presently foresee what a typical planetary landing mission/vehicle combination will be, it is informative to look at various mission/vehicle combinations to gain insight into vehicle and engine size and operating requirements, and also possibly indicate the feasibility of the systems formulated.

Three interplanetary missions, a Mercury soft-landing probe, a Mars roundtrip and a Venus roundtrip, were selected for use in analyses of overall vehicle requirements. Characteristics of these missions are presented in Table 33 . The payloads selected for the roundtrip missions are indicative of manned mission requirements. The vehicles utilized for performance of these missions employed chemical bipropellant rockets (high-energy cryogenic propellants in most instances) for propulsive phases and ablative shields for aerodynamic braking phases.

The resulting vehicles are described in Tables 34 and 35 . The magnitude of interplanetary ventures is clearly demonstrated by the fact that the launch weight is 12,910,000 pounds to satisfy the relatively modest objectives of the Mercury mission and 2,496,000,000 pounds for the manned Venus mission. Although these values do not imply a need for a single vehicle of the indicated weight (rendezvous methods could be employed), it

(A)

(B)

TABLE 32

ENVIRONMENT DESIGN GUIDE

	EARTH*	MOON	VENUS	MARS	MERCURY
ATMOSPHERE	OVER 500,000 FT	None	Approx. 5000,000 ft Thick	Approx. 1,500,000 ft Thick	None or ≈ 45 ft Thick
PRESSURE	P_{SL} = 2116 lb/ft^2	2116 x 10^{-12} lb/ft^2	P_{SL} = 8500 lb/ft^2	P_{SL} = 178 lb/ft^2	Same as Moon
TEMPERATURE	T_{SL} = 519°R	—	T_{SL} = 1270°R	T_{SL} = 490 °R	—
MASS DENSITY	ρ_{SL} = 2.377 x 10^{-3} Slugs/Ft^3	2.377 x 10^{-15} Slugs/Ft^3	ρ_{SL} = 7.5 x 10^{-3} Slugs/Ft^3	ρ_{SL} = 2.1 x 10^{-4} Slugs/Ft^3	Same as Moon
SPEED OF SOUND	a_o = 1116.89 Ft/Sec @ S.L. STD	—	1.01 a_o Earth	0.81 a_o Earth	—
COMPOSITION	21% O_2, 78% N_2, Rare Gases	Possibly Some Unknown Proportions of A, Kr, Xe, SO_2, CO_2, H_2S, O_3	90% CO_2, 10% N_2 By Weight	55% N_2, 5% A By Volume	Possibly CO_2
WIND	YES	None	Windy With Gusts	Up to 20 MPH - Possibly Higher	None
CLOUDS	YES	None	Opaque 58,000 ft to 132,000 ft Cloud Top	White Yellow Blue	None
HUMIDITY; RAIN	YES	None	—	—	None
LIGHTNING	YES	None	Possible	—	None
ECLIPSE EFFECT	YES	Rapid Change of 340 R°	—	—	—
TOPOGRAPHY					
MAPS	MANY AVAILABLE	NAA/SID	NONE	NAA/SID MAP	NONE
SURFACE TERRAIN					
MOUNTAINS, PLAINS, SEAS, VALLEYS	ALL EXIST - VARIABLE SMOOTHNESS	ROUGH, MOUNTAINOUS, CRATERS, PITS, RIDGES	DRY, DUSTY & MOUNTAINOUS	RELATIVELY SMOOTH, ELEVATED AREAS & PLATEAUX, NO MOUNTAINS OVER 9000 FT, DESERTS	ROUGH & IRREGULAR, PARCHED
SURFACE COMPOSITION RIGIDITY OR STRESS	ROCK, WATER, DUST, ETC.	DUST (2-3 CM) ROCKY 250 psi to 10000 PSI CRUSHING STRESS	UNKNOWN	SIMILAR TO LIMONITE (IRON HYDROXIDE) VEGETATION POSSIBLY	CRUSTY LAVA & ROCK
SURFACE TEMPERATURE	350 °R to 600 °R	210 °R to 720 °R	POSSIBLY 1060 °R	366 °R to 516 °R	TWILIGHT ZONE 260 °R to 960 °R
VOLCANOS	YES	POSSIBLE	UNLIKELY	UNLIKELY	—
FLOODS	YES	NO	VERY UNLIKELY	—	NO
EARTHQUAKES	YES	POSSIBLE	UNLIKELY	—	—

*ARDC MODEL ATMOSPHERE NO. 86, 1956

FORM R 18-G

is evident, at least in the latter instance, that propulsive devices more efficient than liquid chemical rockets are required in some phases of the mission.

TABLE 33

SELECTED MISSIONS

Mercury Landing Probe Mission	
Time Mission Departure	10 May 1973
Earth-Mercury Coast Time, days	90
Payload, pounds	2000
Mars Landing and Return Mission	
Time Mission Departure	6 June 1971
Earth-Mars Coast Time, days	80
Mars Stay Time, days	12
Mars-Earth Coast Time, days	260
Total Mission Time, days	352
Payload, pounds	
Venus Landing and Return Mission	
Time Mission Departure	30 November 1966
Earth-Venus Coast Time, days	125
Venus Stay Time, days	5
Venus-Earth Coast Time, days	145
Total Mission Time, days	275
Payload, pounds	50,000

APPENDIX C: ENGINE START TECHNIQUES FOR EXTRATERRESTRIAL LANDING ENGINES

A summary of Appendix C is presented in the Propulsion Design Guide.

TABLE 34

MERCURY LANDING MISSION

Required Maneuvers	Type of System	Vehicle Weight at Maneuver Initiation, pounds
1. Boost-to-Earth Orbit	Alternatives	
	A. Two-Stage Vehicle	12,910,000
	First Stage - O_2/RP	
	Second Stage - O_2/H_2	
	B. 3 Rendezvoused C-5's	18,000,000
	C. 12 F-1 Nova	14,400,000
2. Earth Orbit Departure	One O_2/H_2 Stage	555,000
3. Space Transfer Propulsion		71,000
Attitude Control	Cold Gas System	
Midcourse Corrections	O_2/H_2 System	
Terminal Correction	Performed by Midcourse System	
4. Mercury Orbit Establishment	Two O_2/H_2 Stages	63,840
5. Landing From Orbit	One F_2/H_2 Stage	6,160
6. Hover Translation	Performed by Landing Stage	2,880
7. Mercury Landing Terminal Deceleration Phase	Frangible Tube Landing Legs	2,742
		(Payload = 2,000)

TABLE 35

PLANETARY LANDING AND RETURN MISSIONS

Required Maneuvers	Mars: Type of System	Mars: Vehicle Weight at Maneuver Initiation, pounds	Venus: Type of System	Venus: Vehicle Weight at Maneuver Initiation, pounds
1. Boost-to-Earth Orbit	Alternatives		Alternatives	
	A. Two Stage Vehicle First Stage - O_2/RP Second Stage - O_2/H_2	118,200,000	A. Two Stage Vehicle First Stage - O_2/RP Second Stage - O_2/H_2	2,496,000,000
	B. 9 Rendezvoused 12 F-1 Novas	149,600,000	B. 100 Rendezvoused O_2/H_2 Vehicles Takeoff Thrust = 24,000,000 pounds	2,035,200,000
	C. 5 Rendezvoused O_2/H_2 Vehicles Takeoff Thrust = 24,000,000 pounds	96,000,000		
2. Earth Orbit Departure	One O_2/H_2 Stage	5,090,000	One O_2/H_2 Stage	107,300,000
3. Space Transfer Phase Propulsion Attitude Control Midcourse Correction Terminal Correction	 F_2/H_2 System F_2/H_2 System Performed by Midcourse System	1,311,000	 Cold Gas System F_2/H_2 System Performed by Midcourse System	27,690,000
4. Braking for Planetary Landing	Aerodynamic Entry Vehicle with Ablative Heat Shield	1,098,000	Aerodynamic Entry Vehicle with Ablative Heat Shield	23,560,000
5. Planetary Landing Terminal Deceleration Phase	Pararocket - Frangible Tube Landing Legs	1,003,000	Parachute - Frangible Tube Landing Legs	19,790,000
6. Planetary Takeoff to Orbit	One F_2/H_2 Stage	726,000	Three F_2/H_2 Stages	13,880,000
7. Planetary Orbit Departure	One F_2/H_2 Stage	235,900	One F_2/H_2 Stage	418,000
8. Space Transfer Phase Propulsion Attitude Control Midcourse Corrections Terminal Correction	 Cold Gas System N_2O_4/50-50 System Performed by Midcourse System	85,900	 Cold Gas System N_2O_4/50-50 System Performed by Midcourse System	79,500
9. Braking for Earth Landing	Aerodynamic Entry Vehicle with Ablative Heat Shield	67,240	Aerodynamic Entry Vehicle with Ablative Heat Shield	62,310
10. Earth Landing Terminal	Parachute - Frangible Tube Landing Legs	56,920	Parachute - Frangible Tube Landing Legs	55,240
		(Payload = 50,000)		(Payload = 50,000)

FORM R 18-G-18

APPENDIX D: LANDING GEAR SYSTEMS FOR EXTRATERRESTRIAL LANDING VEHICLES

A design study of landing gear systems for extraterrestrial landing vehicles was performed to define the requirements of touchdown devices, to determine the weights of various landing gear configurations and to select the system best suited to the lunar landing mission. The vehicle considered had a high center of gravity in relation to its base diameter, representative of a vehicle which includes an expended propulsion system. To counteract the resulting tipping instability, it was necessary to place the landing feet at large radial distances from the vehicle, thus making the structural weight of the landing gear much heavier than that of the energy absorber. Thus, the energy absorbed per unit mass is not the most important factor in selecting the energy absorber, and other considerations such as packaging and reuseability can be considered to be equally or more important.

The basic single and dual tripod configurations considered are illustrated in Figures 19 and 20 . Design of various landing gear systems indicated that the total weight of the system ranges between 3 and 9 percent of the gross weight of the vehicle at touchdown. Increasing or decreasing the impact velocity causes corresponding changes in the landing gear weight. Three different energy absorbers, the hydraulic cylinder, frangible tube, and crushable metal honeycomb were considered so that the effect of varying the energy absorber on total landing gear weight could be assessed. Results indicated that for long stroke landings (i.e., at high impact velocities and low deceleration rates) the frangible tube is superior to the hydraulic cylinder. For shorter stroke applications, the two are approximately equal, the hydraulic cylinder having the advantage of repeated landing capability.

APPENDIX E: LUNAR SURFACE STORAGE OF LIQUID PROPELLANTS

A preliminary investigation of the storability characteristics of oxygen/hydrogen (O_2/H_2) and fluorine/hydrogen (F_2/H_2) propellant systems at a lunar equatorial site has been conducted to determine potential storage problems and indicate storage system weight requirements.

During lunar surface residence, heat transfer to the propellants from the Sun, the moon, and from components of the rocket vehicle will result in a propellant pressure and temperature rise for a nonvented system, or propellant boiloff for a vented system. Investigation of a pump-fed propulsion system using nonvented tanks, because of their greater simplicity, was made. For the nonvented storage system, the propellant tanks are sealed and the absorbed heat causes an increase in pressure and temperature of the propellants, and thus, thermal protection of the cryogenic propellants from the adverse heating environment of the moon is required. If the thermal protection requirements are extensive, the potential advantage normally

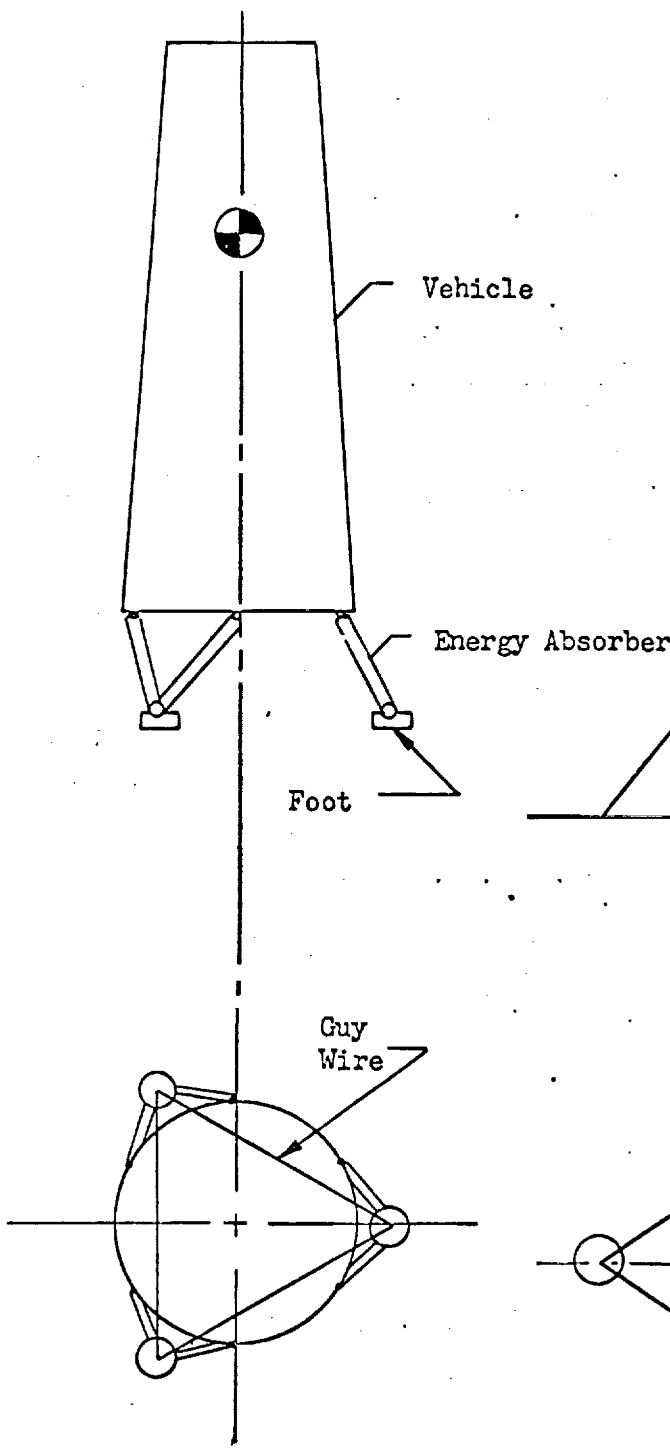


Figure 19 Schematic Representation of Landing Vehicle With Tripod Landing Gear

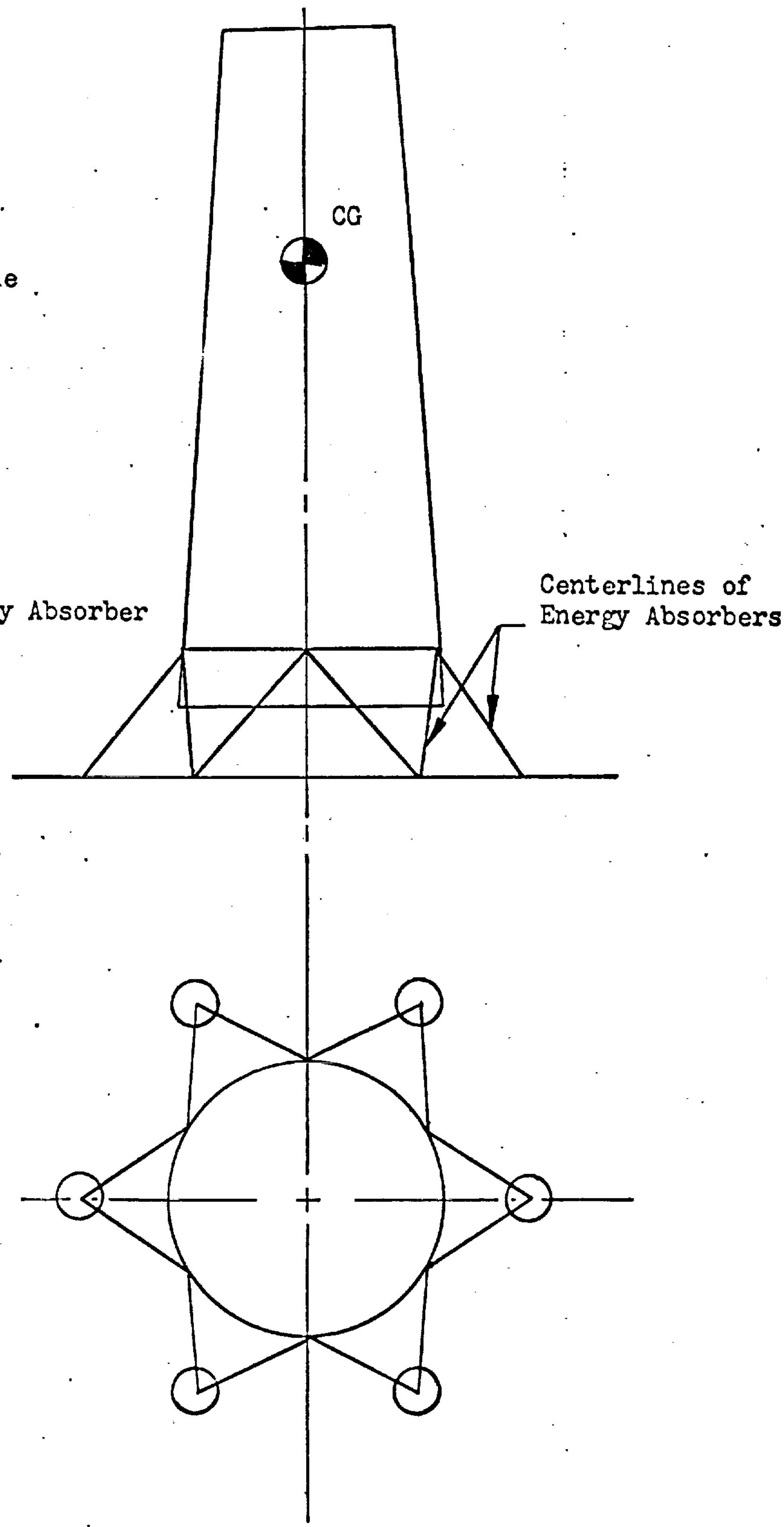


Figure 20 Schematic Representation of Landing Vehicle with Dual Tripod Landing Gear

associated with the use of high energy cryogenic propellants is not obtained. Propellant tank design pressures (dictated by tank weight allowances) must not be exceeded, and the propellant must be maintained at a temperature (and vapor pressure) low enough for engine operation.

A lunar vehicle having a gross weight of 122,000 pounds, a total propellant weight of 61,800 pounds, and payload weight of 50,000 pounds was considered. These weights would provide an ideal velocity increment of approximately 10,000 ft/sec (for a lunar takeoff mission), based on a propellant fraction of 0.86, and a propellant specific impulse of 440 seconds. For the vehicle system investigated, the fuel and oxidizer are considered to be contained in separate cylindrical tanks of equal diameter with 2:1 ellipsoidal bulkheads, a combined length to diameter ratio of 1.75, and located below and in line with the payload. The vehicle was assumed to be sitting upright on the lunar equator, and no shadow shielding except that provided by the payload capsule was considered; therefore, the conditions assumed are severe with respect to storability problems.

By the results it was indicated that storage (1 or 2 lunar cycles) of a cryogenic liquid-propellant vehicle system on the lunar surface appears feasible for a nonvented storage system using superinsulations, surface coatings, and radiation shields. The main factor, other than insulation properties, is the allowable propellant pressure rise. Storage weight penalties up to approximately 2300 pounds (Δp = 20 psi) for a 122,000 pound gross weight (50,000 pound payload) vehicle are experienced. The F_2/H_2 propellant combination is more readily storable than O_2/H_2 (based on storage penalty weight) for the conditions assumed in this study. This is due mainly to the larger fuel tank dimensions (and therefore a greater tank insulation weight) for the O_2/H_2 system brought about by its lower mixture ratio.

Several of the basic assumptions used in this preliminary study significantly influenced the results obtained. For example, a less pessimistic assumption on the latitude location of the vehicle (assumed at the lunar equator), and some shadow shielding schemes could reduce the storage penalty weights of the cryogenic systems considerably. The heat transfer analysis is simplified; calculations are based on an average tank skin temperature; however, if instantaneous skin temperatures completely penetrate the insulation, a corresponding fluctuation in tank pressure rise would result. Storage time would then be decreased since the pressure peaks would dictate maximum allowable storage pressure instead of average pressure. Changes in assumptions of propellant pressures, vehicle design characteristics, and the lunar mission would also undoubtedly influence the results of the study.

The results indicate that the storage penalty weights range up to 2 percent of vehicle gross weight (or approximately 4.6 percent of payload weight); thus, the effect of lunar storage for 1 or 2 cycles (up to approximately 60 days) appears significant but does not prohibit use of high energy cryogenic liquid propellants or negate their performance advantage.

PROPULSION DESIGN GUIDE

INTRODUCTION

The Propulsion System Design Guide was compiled to provide a description of the characteristics of optimum propulsion systems for the landing (and in some instances, takeoff) phases of interplanetary missions. Primary emphasis was placed on the presentation of optimum operating parameters (e.g., chamber pressure, expansion area ratio) and the effect of variation of these parameters on vehicle payload capability. The selection of other system characteristics, in particular, subsystem configurations, was reviewed briefly; a detailed investigation of this aspect of the design of propulsion systems for interplanetary missions is currently in progress in a related study under NASA Contract NAS 7-164, Optimization of Operating Conditions for Manned Spacecraft Engines.

The data presented in this section offer a useful insight into the effects of various system parameters on one another and on vehicle payload capabilities for the missions considered. The stated values are correct, however, only for systems whose configurations and performance are in agreement with the assumptions stated in the "Engine Parameter Optimization" section of Volume 2B, and which perform thier required maneuvers in accordance with the trajectory techniques described elsewhere in Volumes 2A and 2B. The importance of this restriction can be emphasized, for example, by the fact that the assumption regarding whether or not interstage structure is designed only for structural adequacy or to serve as a meteorite shield has a factor-of-two effect on optimum expansion area ratio. Recognition of the interrelationship between this Propulsion System Design Guide and the detailed analyses upon which it is based is therefore essential to proper use of the data contained herein.

PROPULSION SYSTEM CHARACTERISTICS

Definition of a propulsion system includes selection of quantitative characteristics such as velocity requirement, thrust-to-weight ratio, throttling ratio, chamber pressure, and expansion area ratio, and qualitative features such as propellant combination, feed system type, nozzle type and arrangement, start method, and thrust vector control technique. With the exception of engine start (which is treated in some detail in Appendix C of Volume 3) the study, Propulsion Requirements for Soft Landing in Extraterrestrial Environments, was concerned with quantitative aspects of landing propulsion engines; Qualitative features were selected on the basis of available propulsion information, and in some cases alternative systems (e.g., pump- and pressure-fed) were considered. The following features were selected for the optimization of propulsion system characteristics:

1. Pump-fed systems for possible manned 50,000-pound and 500,000-pound thrust vehicles and pressure-fed systems for unmanned 5,000-pound thrust applications.

2. Regeneratively cooled 80-percent bell nozzles; fully cooled for pump-fed systems and cooled to an expansion area ratio of 20:1, with ablative cooling thereafter, for pressure-fed systems.

3. Single nozzle configurations

A few comments regarding propellant selection, thrust vector control and engine start are presented below.

Propellant Selection

A wide variety of propellants is available for use in chemical rocket propulsion systems; however, because of factors such as specific impulse, availability, toxicity, etc., the number of combinations that can actually be considered for application to the major propulsive phases of space missions is relatively limited. A comprehensive presentation and evaluation of potential liquid bipropellant candidates, ranging from conventional current propellants to exotic high energy combinations, is provided in Reference (3).

For each of the missions considered in this document, propulsion systems employing three different propellant combinations, representing a broad range of performance and logistic characteristics, were analyzed. Nitrogen tetroxide (NTO) with an equal weight mixture of hydrazine and UDMH (termed 50-50) provides payload capability comparable to LOX/RP-1 or an advanced relatively high performance solid propellant; but it offers hypergolic ignition and favorable space storage qualities, advantages which can, in some instances, over-ride the low specific impulse. Liquid Oxygen/Hydrogen represents a class of propellants with substantially higher specific impulse than NTO/50-50; it does, however, lose some of its advantage because of the low density and extremely low temperature of hydrogen, characteristics which translate to propellant fractions lower than are obtained in other systems. Liquid Oxygen/Hydrogen, in fact, displays payload capability for some missions similar in magnitude to another potential high-energy combination, Fluorine/Hydrazine, despite a 30-second vacuum specific impulse advantage for O_2/H_2. Fluorine/Hydrogen approaches the ultimate in liquid bipropellant systems. In addition to high specific impulse, F_2/H_2 optimizes at sufficiently high mixture ratios to suppress the influence of low hydrogen density.

These propellants represent the most likely choices for application to near-future space missions. Beyond F_2/H_2, certain bipropellant and tripropellant combinations offer even greater payload capabilities; their use, however, is probably precluded by the impending entry of nuclear rockets into the space propulsion inventory.

Thrust Vector Control

The magnitude of thrust vector control required for performance of space vehicle stage separation operations or rotation maneuvers is generally quite small; in fact, it usually amounts to between a few tenths of one degree and slightly in excess of one degree, and is of the same order of magnitude as the gimbaling capability required to correct allowable engine/vehicle misalignment errors (generally specified as 0.5 degree). This result has been documented adequately to preclude a need for additional discussion here (see, for example, Reference 4). As a result, a 2-degree gimbal displacement capability is adequate, with high statistical confidence, to satisfy the thrust vector control requirements in any landing propulsion system.

Use of the landing propulsion system for translation maneuvers imposes somewhat different thrust vector control requirements. In this instance, magnitude of thrust vector angular displacement affects the duration of, and therefore the propellant requirement for, a given translation maneuver. The time required, as a function of gimbal angle, for a selected vehicle to perform a 45-degree rotation is presented in Figure 21 . As an example, consider the middle curve of Figure 21 ; for the nominal 2-degree gimbal angle selected previously, the rotation takes 1.8 seconds longer than for a 6-degree gimbal angle. For a lunar landing, this difference is equivalent to approximately a 20 ft/sec penalty on the velocity requirements of the system with 2-degree capability. Evaluation of the trade-off between development of a higher gimbal angle system and acceptance of a propellant weight penalty is associated with the details of the final vehicle configuration and the design flight trajectory, and thus is beyond the scope of the present discussion.

The gimbal angle requirement for a space propulsion system is greatest when specifications demand that the engines of a clustered system be capable of compensating for the unbalance created by failure of one or more members of the engine group. The numerous ramifications of engine-out operation extend far beyond gimbal angle restrictions; a comprehensive discussion of the concept is presented in Reference 5 . The gimbal angle requirements for various clusters of uncanted rocket engines are summarized in Figure 22 , taken from the reference.

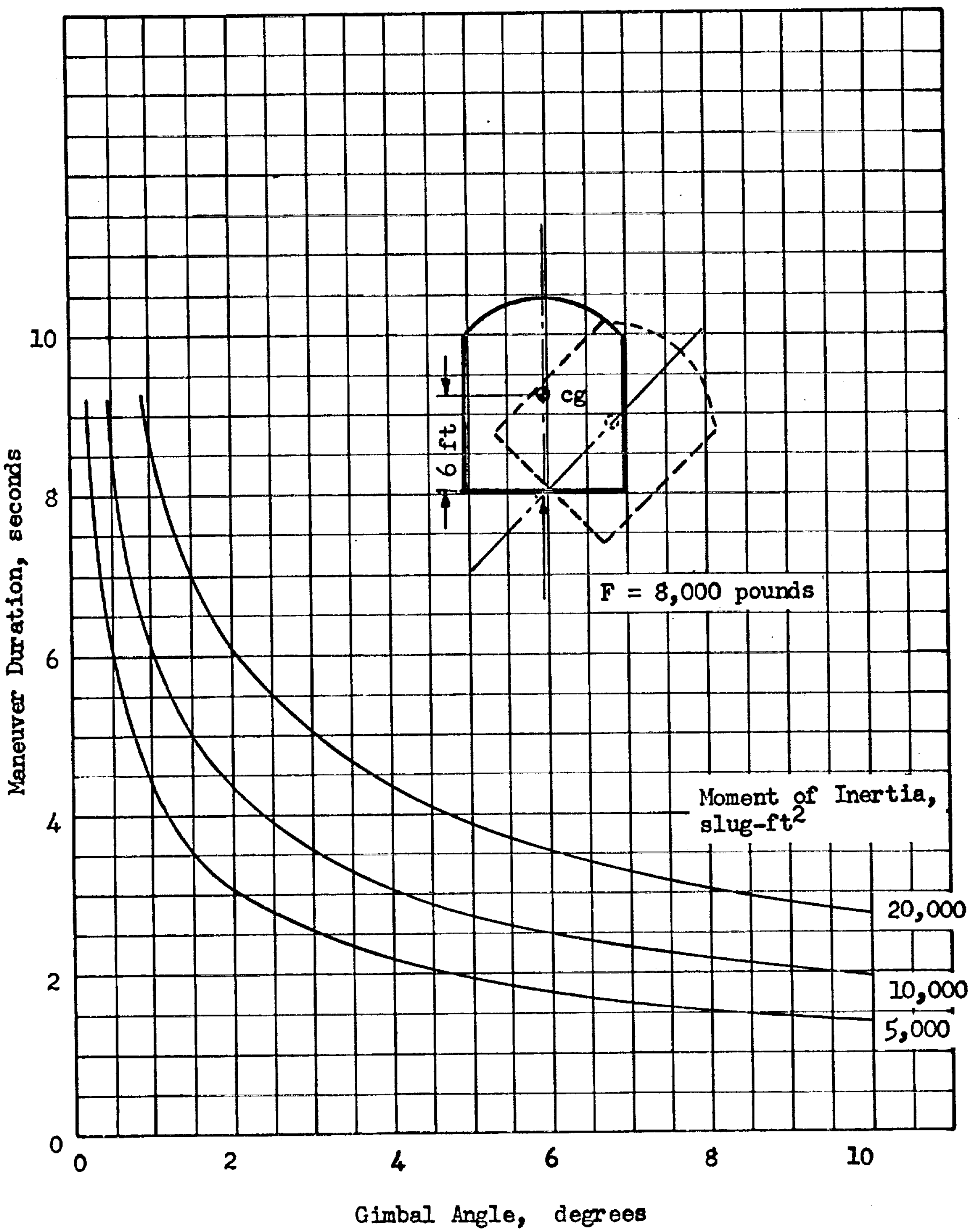


Figure 21 . Time Required for 45-degree Rotation

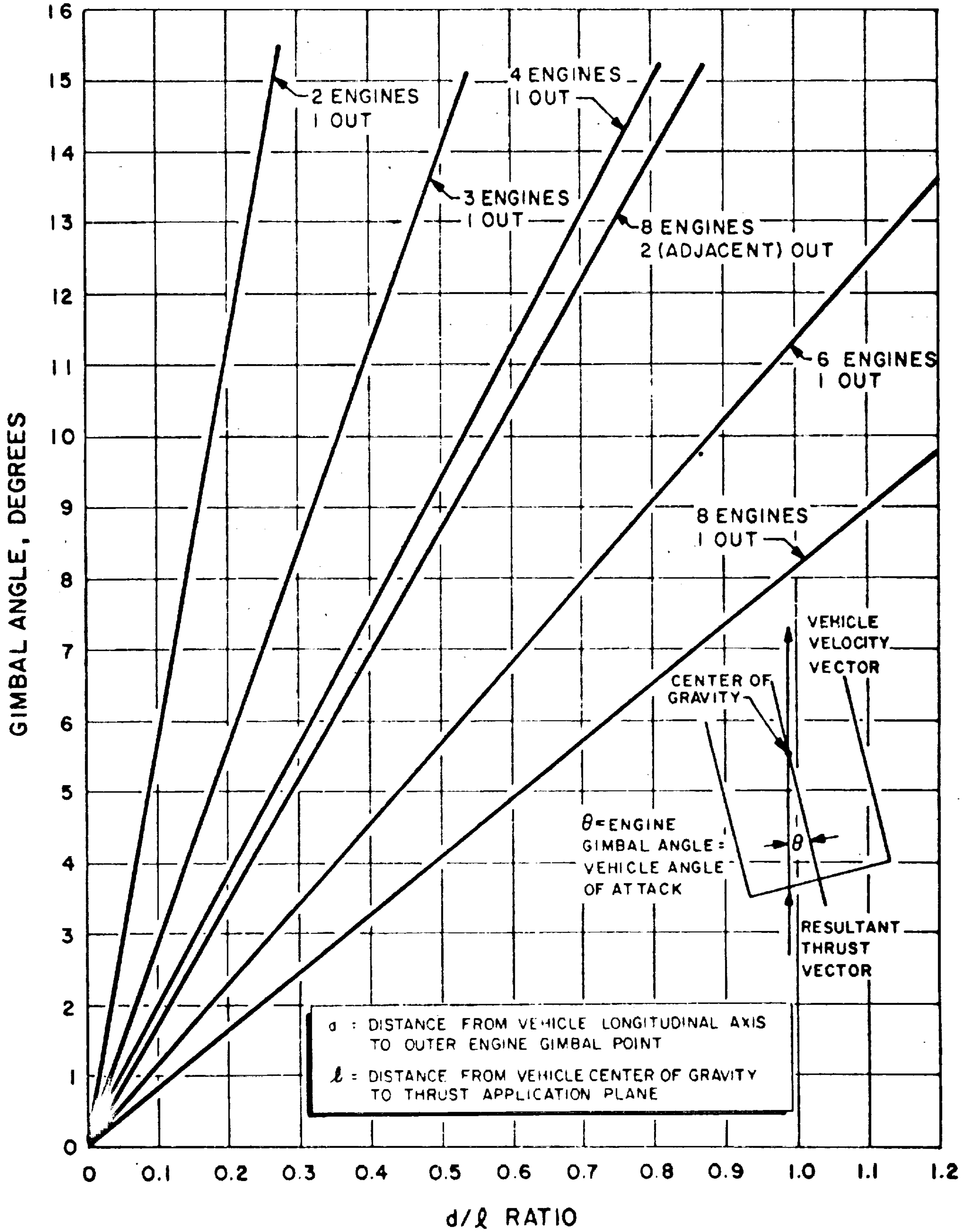


Figure 22. Supplementary Gimbal Requirements for Engine-Out Operation

Two significant differences between landing vehicles and other space vehicles affect the gimbal angle requirements imposed by engine-out operation; both effects are detrimental to the use of the engine-out concept in a landing stage. First, the gimbal angle represents an angle-of-attack during the landing maneuver and therefore jeopardizes the touchdown stability of the vehicle. Second, landing vehicle shape is generally governed by factors other than engine-out operation, and the prevalent low length/diameter vehicles, which offer favorable touchdown stability characteristics, require large gimbal angles to satisfy the restrictions imposed by engine-out operation. For example, a vehicle having vehicle characteristics similar to preliminary designs for a lunar lander powered by a six engine cluster, requires a 12-degree gimbal capability. It appears, therefore, that the engine-out concept cannot be easily employed for extraterrestrial landing vehicles.

The factors considered above lead to the conclusion that 2 degrees is an adequate magnitude of thrust vector control, although additional capability, if it is not difficult or expensive to obtain, offers some advantage for translation maneuvers and provides a margin for meeting unforeseen contingencies. It is important to note that the use of the term, "gimbal," rather than a more general term such as "thrust vector deflection" does not necessarily imply a preference for this technique over others such as secondary fluid injection. The record of reliability demonstrated by gimbaling in liquid propellant systems stands strongly in its favor (particularly for near-future applications), but since a comparison with alternative techniques was not undertaken in the present effort, a definitive selection is not warranted.

Start Systems

The requirement that extraterrestrial landing propulsion systems must start, at least once and several times in most instances, in the zero-g, vacuum environment of space demands that the start system selected must be extremely reliable. An investigation of possible techniques for starting rocket engines under the conditions existing in space was conducted to determine the methods best suited to the task.

The basic requirements of a start system are the supply of propellant to the engine, preparation (e.g., chill-down or controls checkout), supply of turbine power (for pump-fed systems) and ignition of propellants. The latter requirement is automatically satisfied by the use of hypergolic propellants; hypergolicity can be either natural (e.g., NTO/H_3) or induced by chemical additives (usually fluorine or a fluorine compound). The most favorable alternative to hypergolic propellants is the use of a direct spark igniter.

The basic solution to the problem of propellant supply stems from the fact that if the main propellant tanks are pressurized and the main valves opened, the propellant and/or pressurant gas flow, regardless of whether the mixture is combustible or not, will create thrust (for tank settling) and provide liquid propellants to the engine relatively quickly. The indeterminate nature of the transient mixture ratio has little significance for ablative or radiation-cooled nozzle, and represents a problem which can be overcome by suitable design of regeneratively cooled nozzles. The problem of propellant supply can be further alleviated by the use of various surface-tension devices to assure that liquid propellants are initially delivered to the engine inlets.

OPERATING PARAMETERS

The following data summarize the principal characteristics of optimum propulsion systems for several important propulsive maneuvers associated with extraterrestrial soft landings. A summary chart is presented for each propulsion maneuver considered. Optimum characteristics for the propulsion system applicable to performance of the indicated maneuver are presented. Figures and charts specifically related to the maneuver follow each summary chart. The general purpose of these is to provide useful supplementary parametric information on the propulsion requirements for the maneuver.

The assumptions related to the stated parameters are described in Volumes 2A and 2B. In those instances for which a range of number of starts is indicated, the restarts are required if the propulsion system is utilized for one or two terminal corrections in addition to the stated maneuver.

Planet: Earth

Maneuver: Deceleration from 48,000 ft/sec Hyperbolic Arrival Velocity to Optimum Atmosphere Entry Velocity.

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) -/0.32

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) -/4270

Total Starts 1 - 3

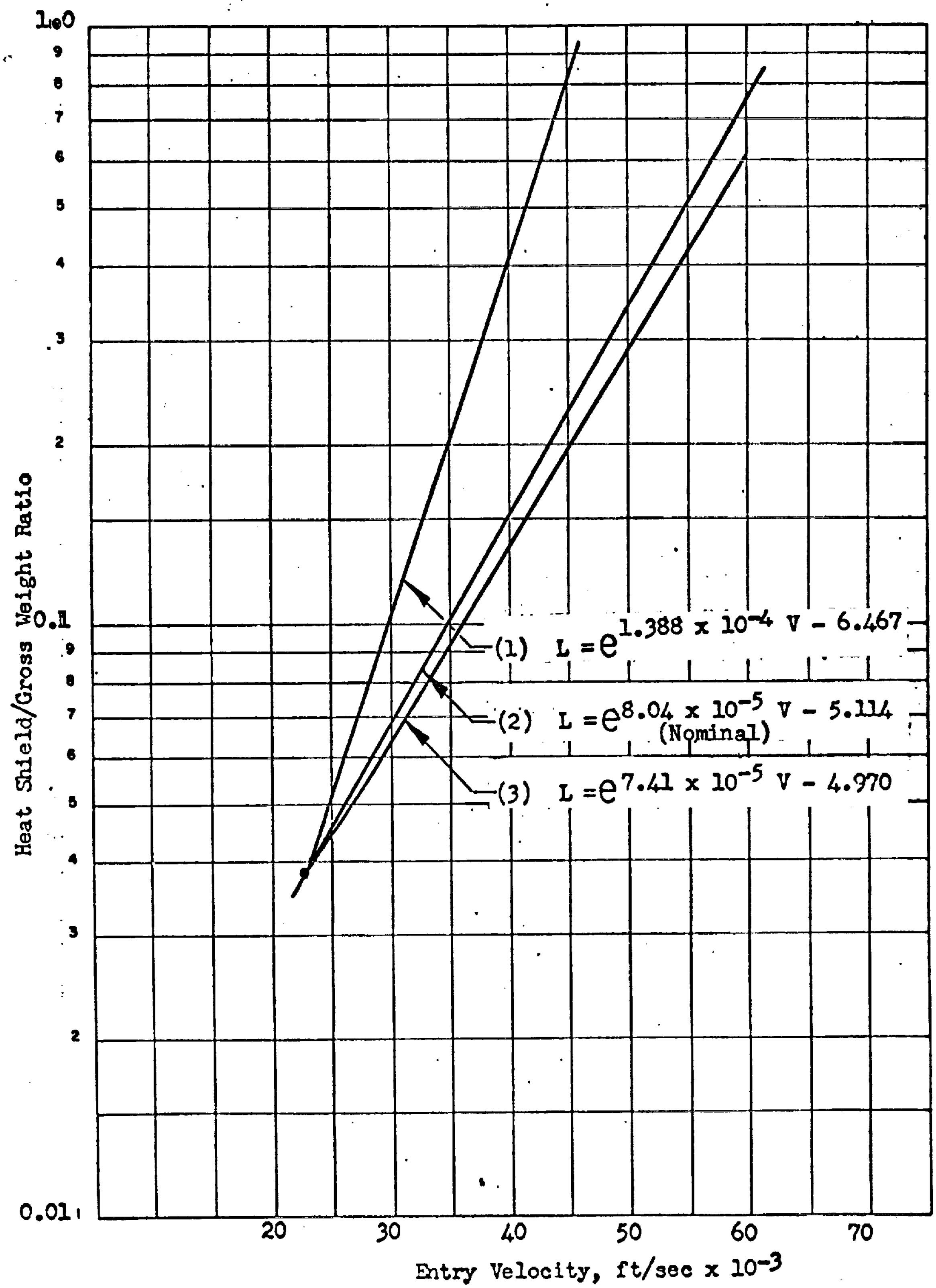
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	80	1350	1400
Expansion Area Ratio	190	330	180
Mixture Ratio	6.1	6.7	6.3
F_2/H_2 System			
Chamber Pressure, psia	110	1740	2090
Expansion Area Ratio	200	270	190
Mixture Ratio	16.9	17.2	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	--	--	--
Expansion Area Ratio			
Mixture Ratio			

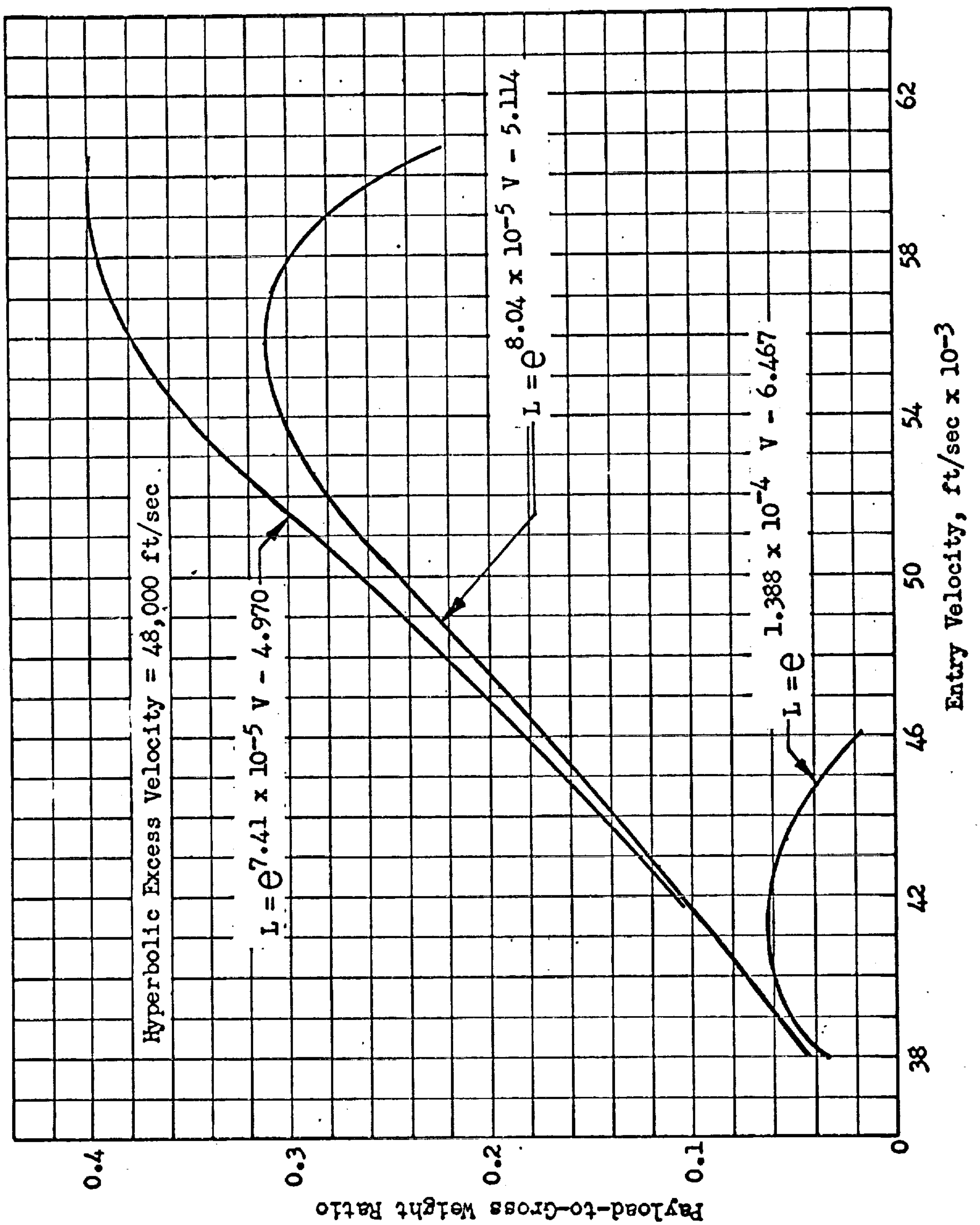
Related Figures

1. Assumed Ablation Shield Weight Characteristic
2. Payload vs Entry Velocity
3. Optimum Entry Velocity vs Hyperbolic Arrival Velocity and Ablation Shield Weight Characteristic

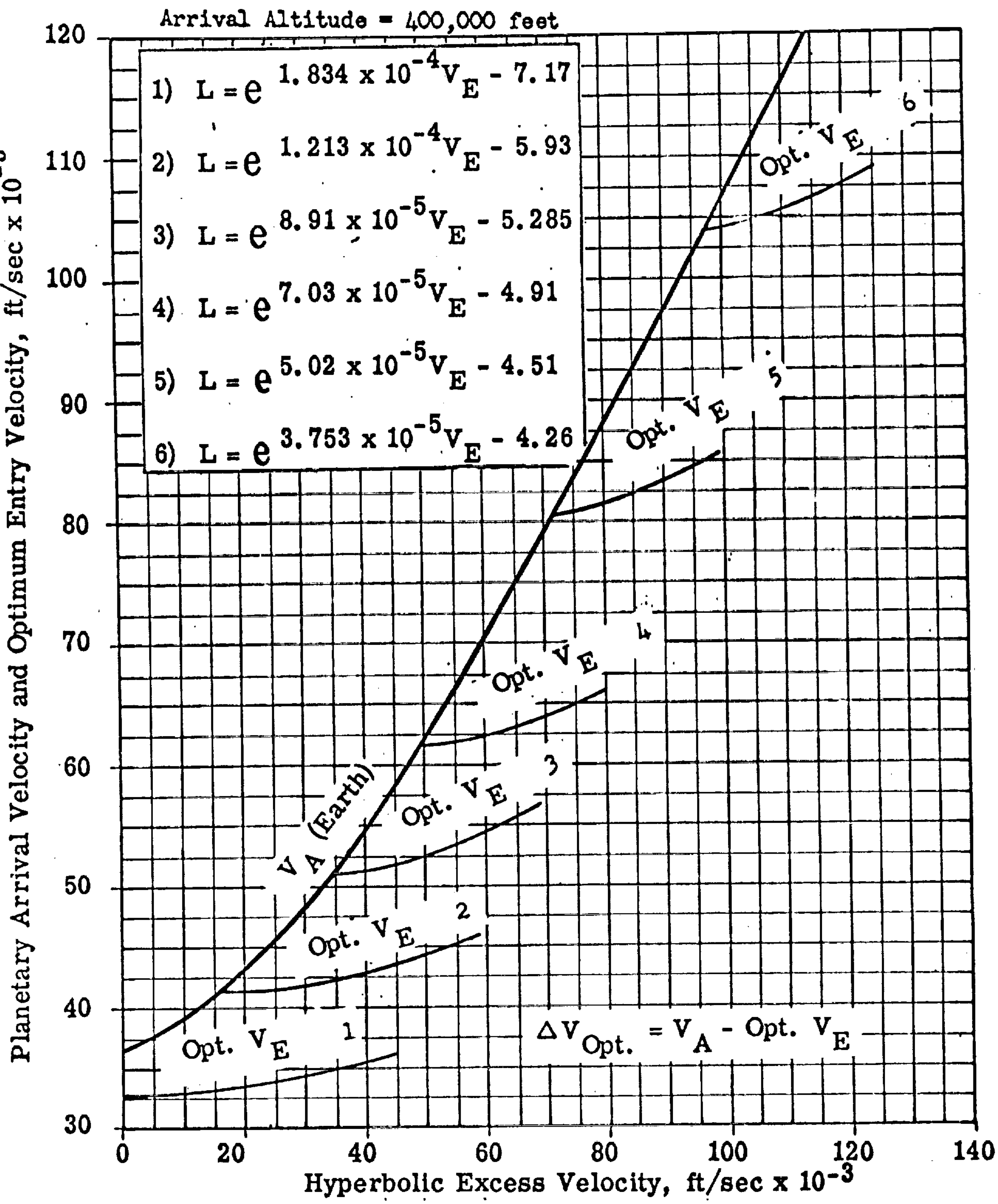
* Based on nonredundant engine system



• Heat Shield Characteristics for Earth Entry Vehicles.



• Propulsion/Ablation Systems for Earth Reentry



Optimum Propulsive ΔV (Impulsive) for Earth Re-Entry of Propulsive/Aerodynamic Systems.

Planet: Earth

Maneuver: 300 n. miles Orbit Establishment from 15,000 ft/sec Hyperbolic Excess Velocity

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) 0.38/0.40

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 13,590/13,580

Total Starts 1 - 3

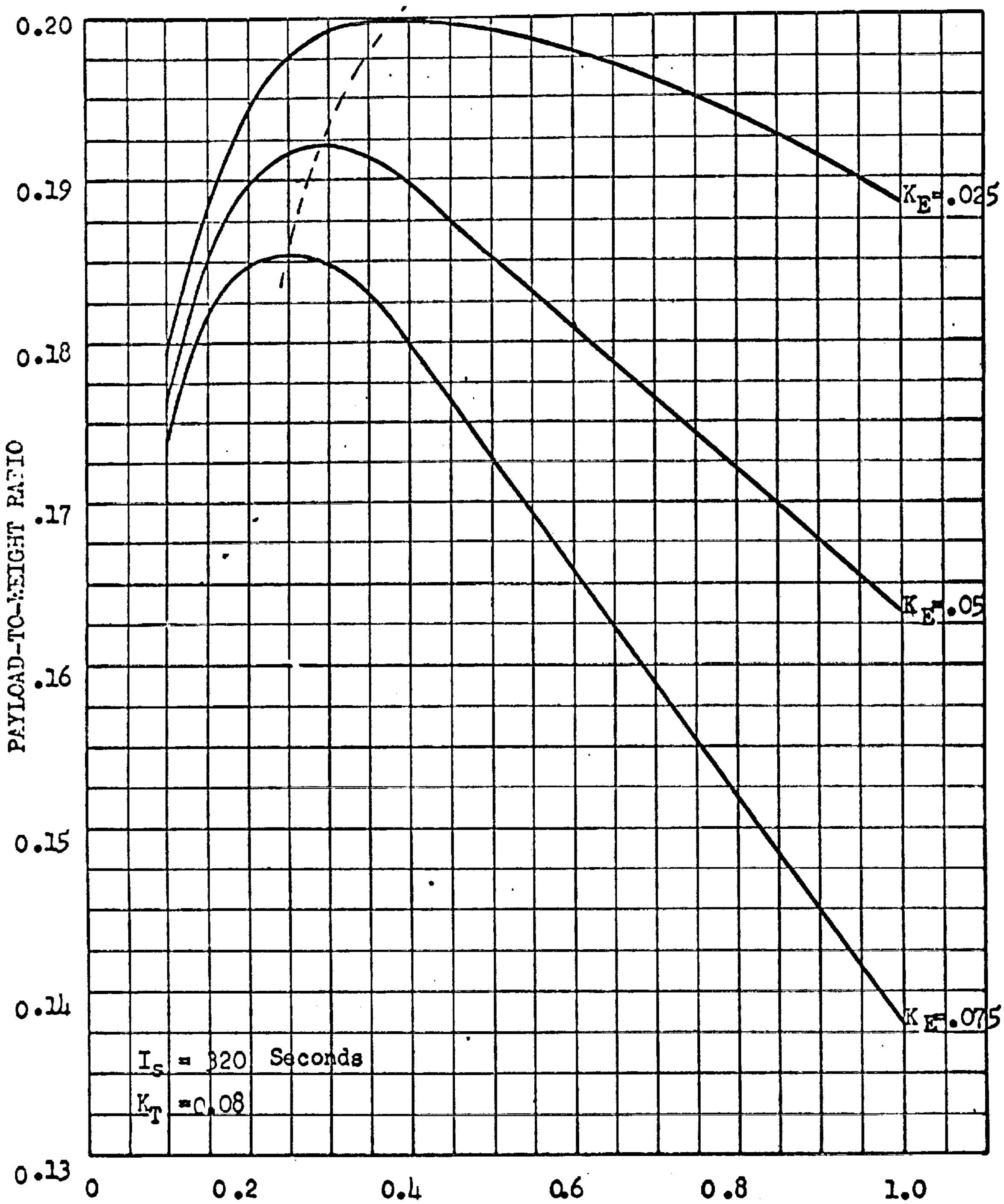
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	60	1320	1280
Expansion Area Ratio	180	390	190
Mixture Ratio	6.2	6.9	6.4
F_2/H_2 System			
Chamber Pressure, psia	85	1730	1830
Expansion Area Ratio	190	360	190
Mixture Ratio	16.9	17.4	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	135	1790	1980
Expansion Area Ratio	290	430	280
Mixture Ratio	2.2	2.2	2.2

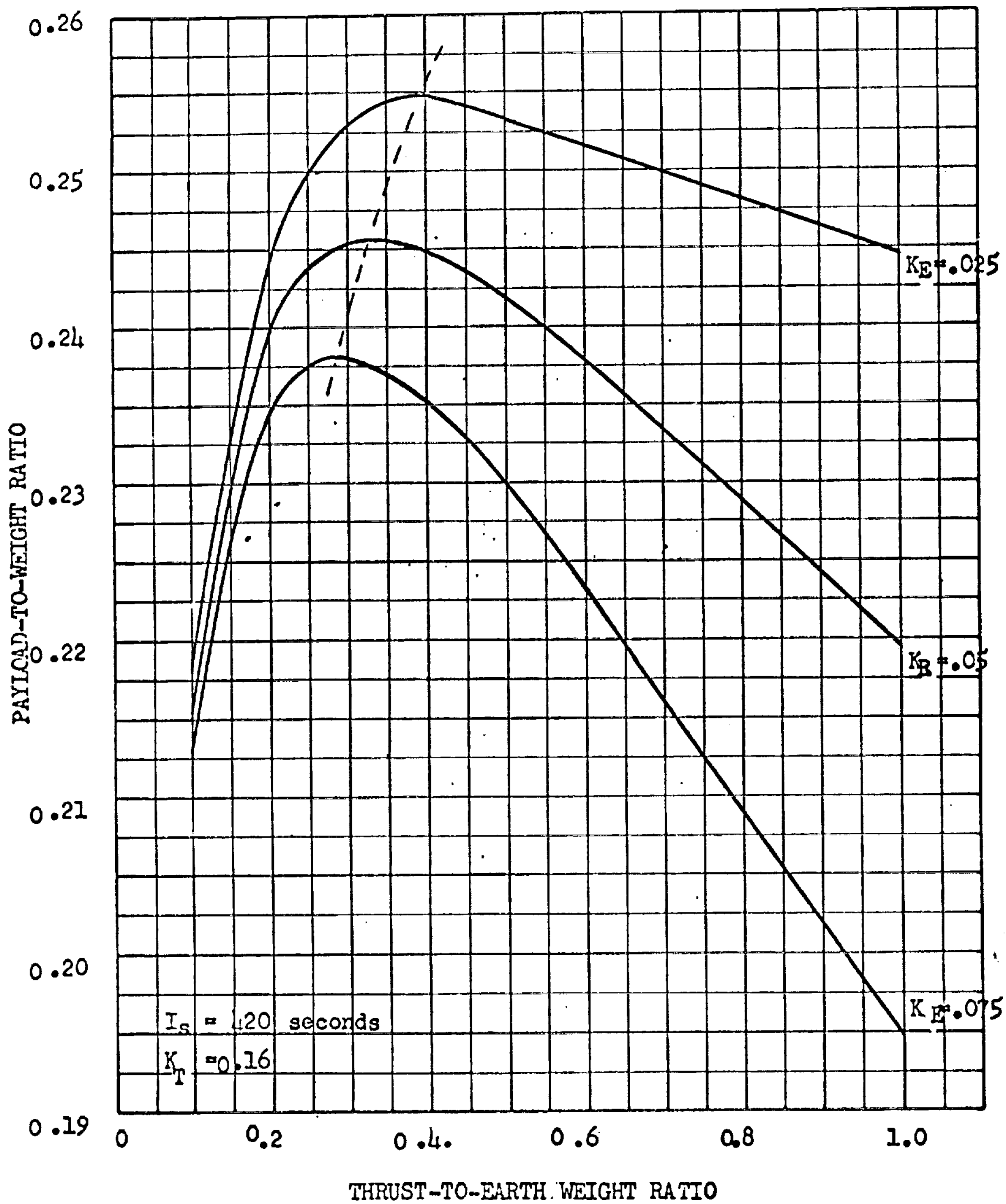
Related Figures

1. Payload to Weight Ratio vs F/W; non-cryogenic
2. Payload to Weight Ratio vs F/W; cryogenic
3. Payload to Weight Ratio vs V_H and λ_p
4. F/W Variation for one Percent Change in Payload

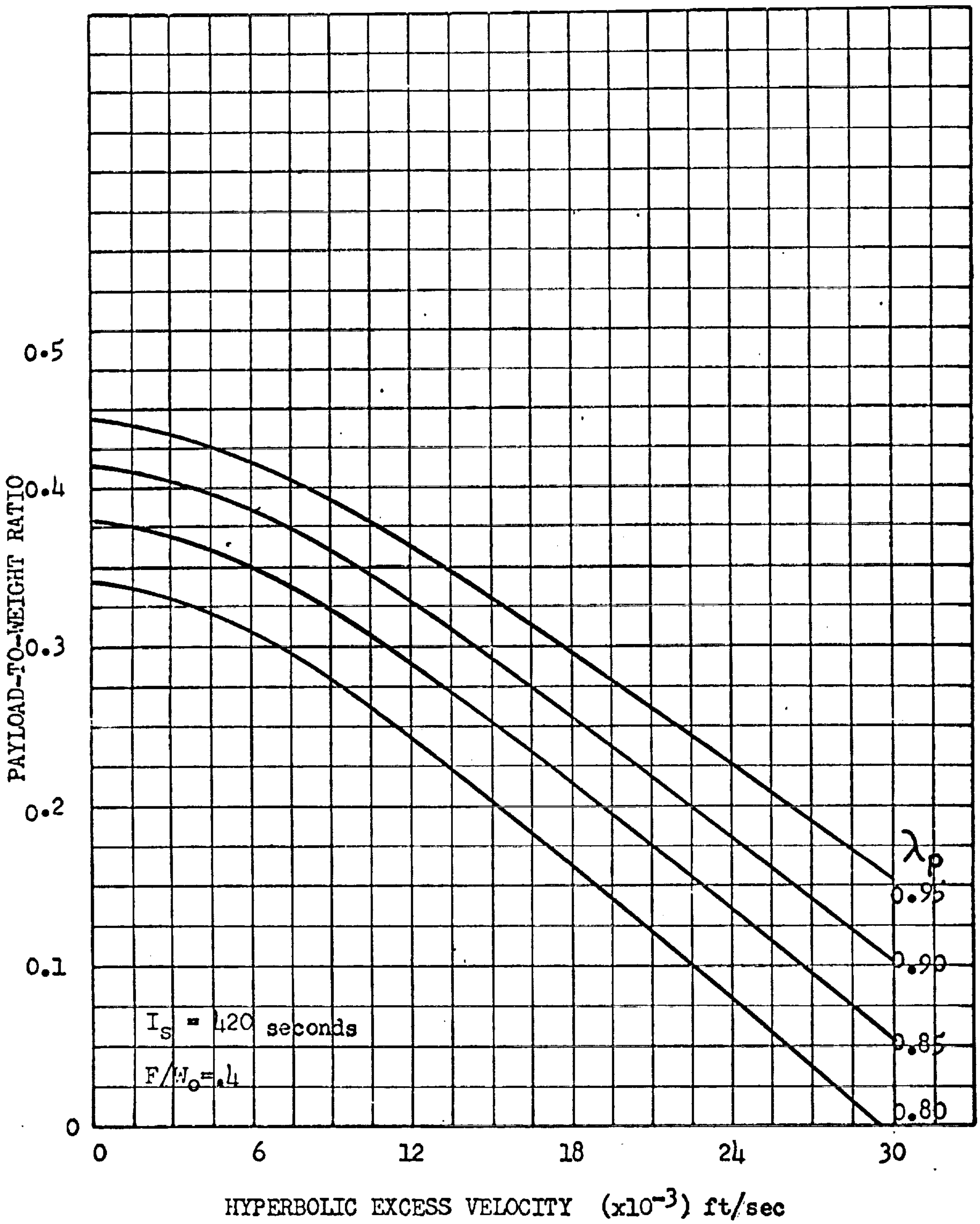
* Based on nonredundant engine system



EARTH - HYPERBOLIC EXCESS = 15,000 ft/sec

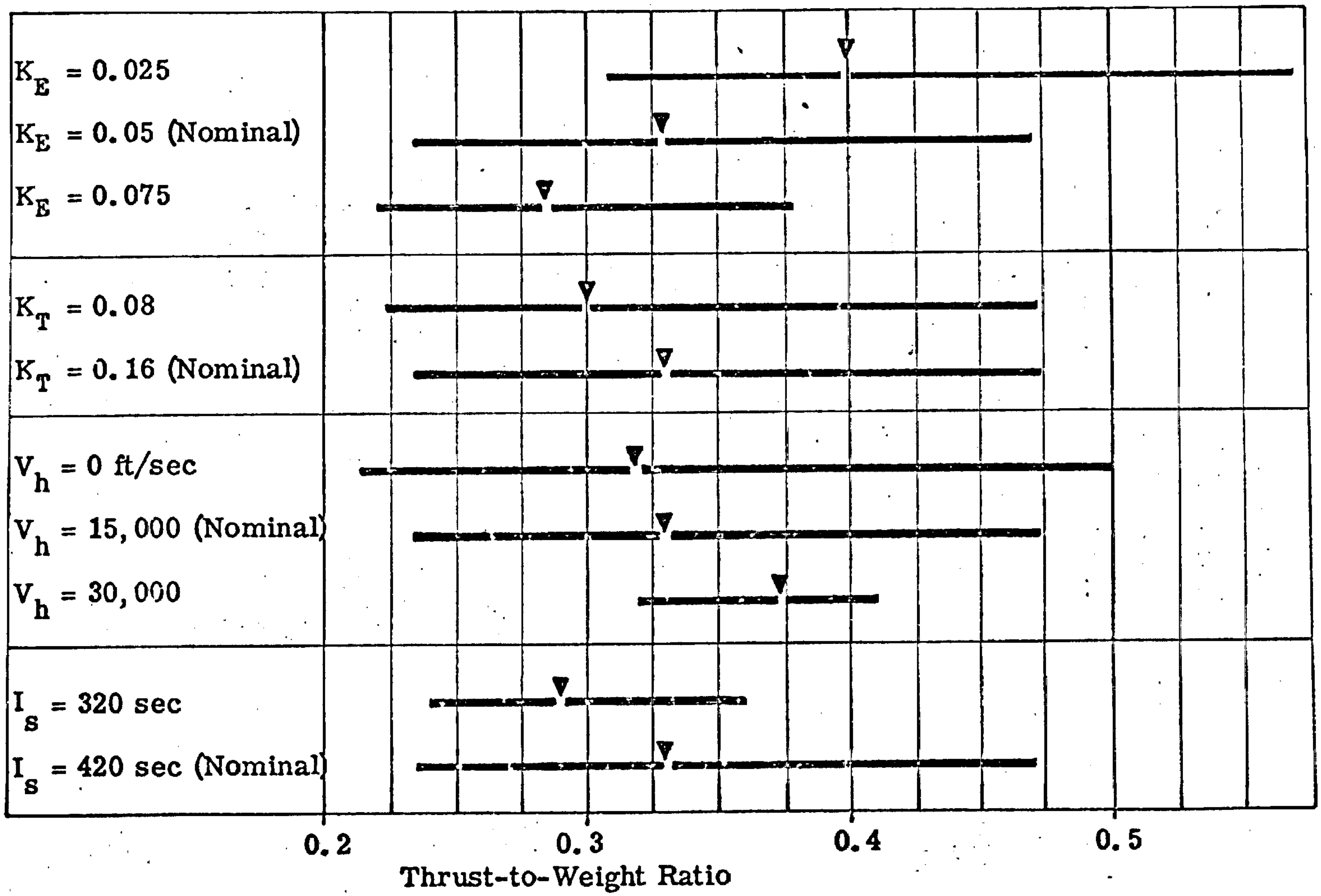


EARTH - HYPERBOLIC EXCESS = 15,000 ft/sec



EARTH PAYLOAD VS. HYPERBOLIC EXCESS

Nominal Vehicle: I_s = 420 sec, K_P = 0.16, K_T = 0.05, V_h = 15,000 ft/sec



Thrust-to-Weight Ratio Variation for 1 Percent Change in Payload for Earth Orbit

Planet: Mars

Maneuver: 300 n. miles Orbit Establishment from 12,000 feet/second Hyperbolic Excess Velocity

Optimum Thrust-to-(Mars) Weight Ratio (non-cryogenic/cryogenic) * 0.38/0.43

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 8,670/8,960

Total Starts 1 - 3

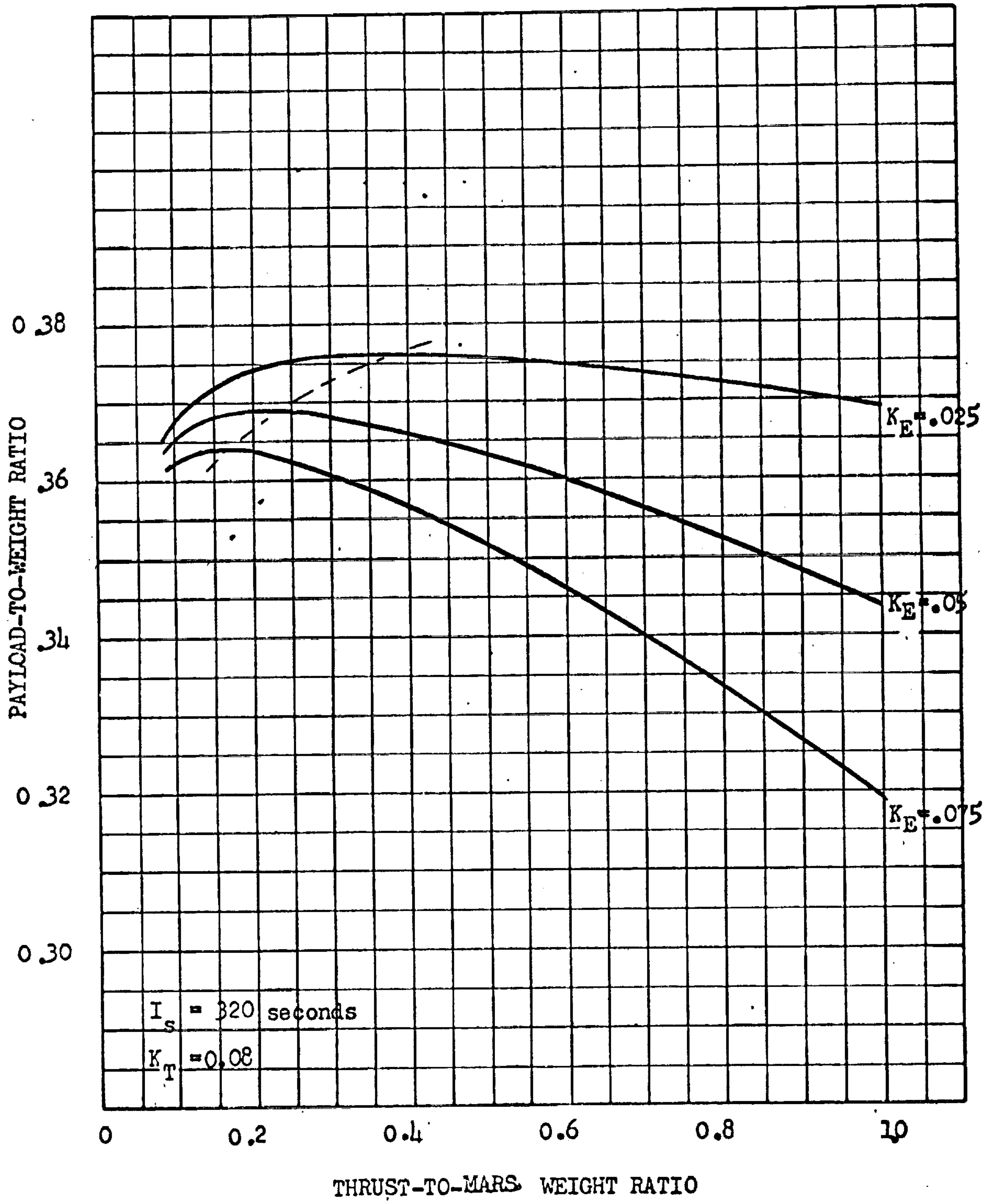
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	65	1420	1390
Expansion Area Ratio	220	450	210
Mixture Ratio	6.2	6.8	6.4
F_2/H_2 System			
Chamber Pressure, psia	85	1800	2030
Expansion Area Ratio	210	380	210
Mixture Ratio	16.9	17.3	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	135	1860	2060
Expansion Area Ratio	350	500	310
Mixture Ratio	2.2	2.2	2.2

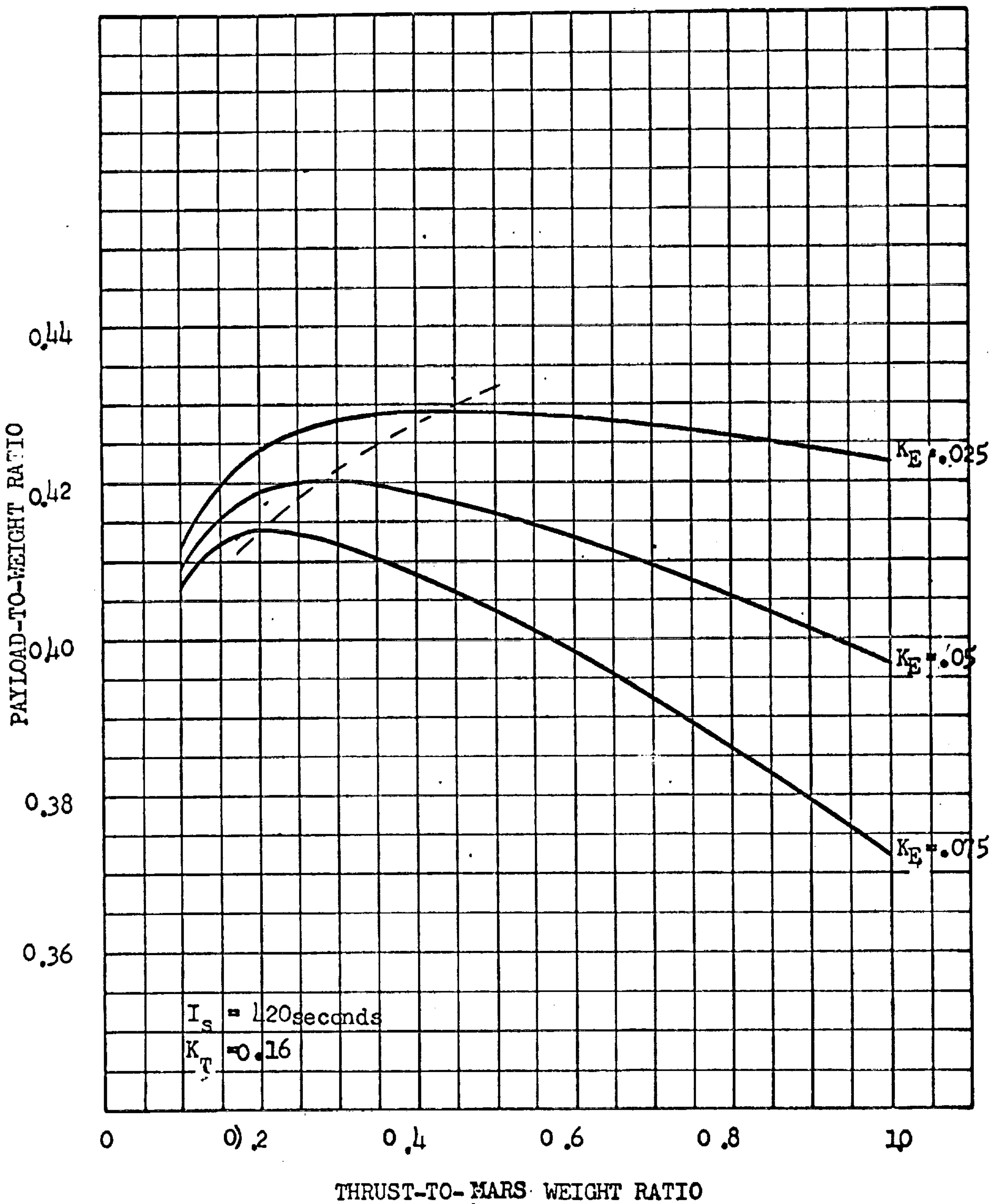
Related Figures

1. Payload to Gross Weight vs F/W; Non-cryogenic
2. Payload to Gross Weight vs F/W, Cryogenic
3. Payload to Gross Weight vs V_H and λ_p; cryogenic

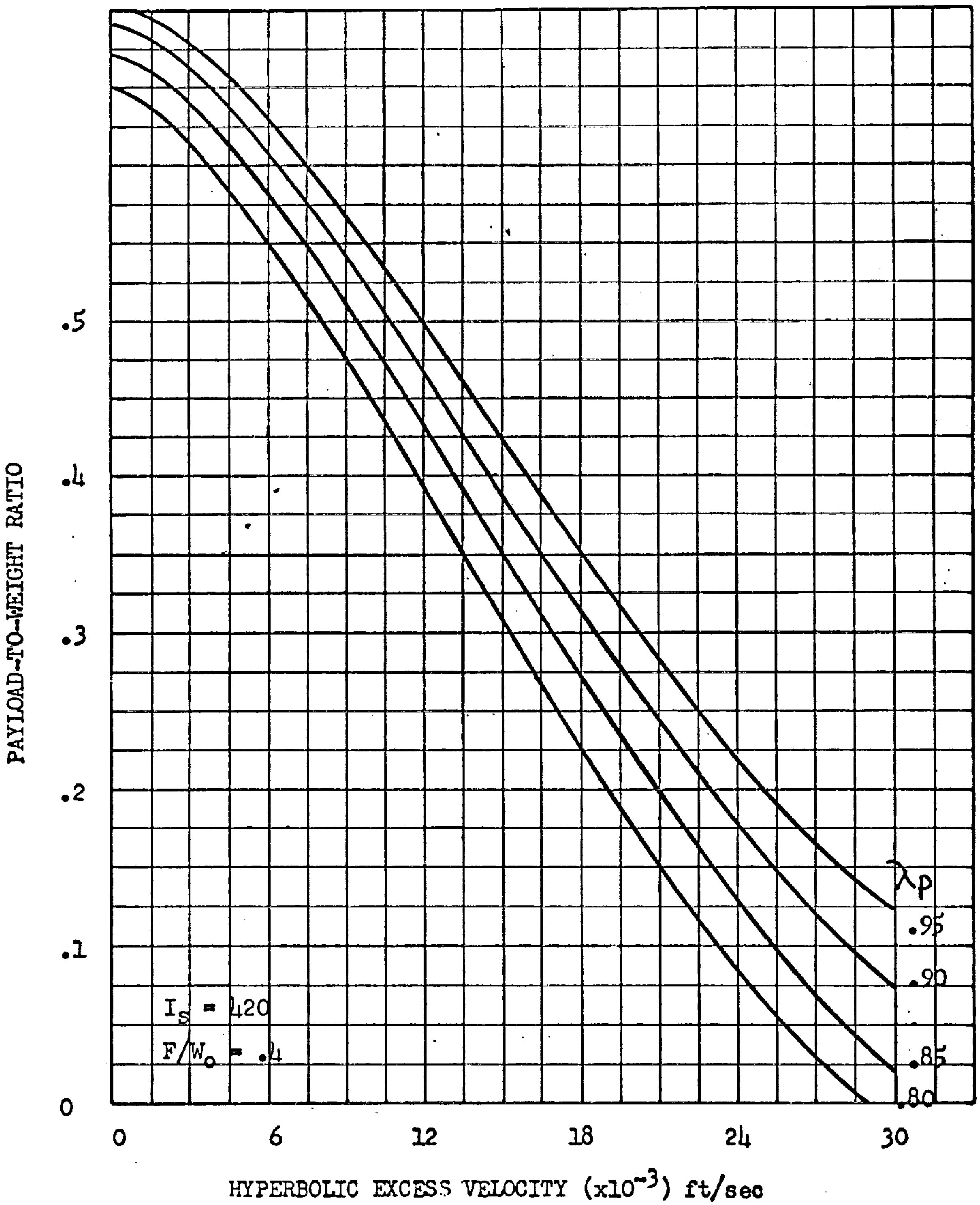
* Based on nonredundant engine system



MARS - HYPERBOLIC EXCESS = 12,000 ft/sec



MARS - HYPERBOLIC EXCESS = 12,000 ft/sec



MARS PAYLOAD VS. HYPERBOLIC EXCESS

Planet: Mars

Maneuver: Take-off to 300 N. Mi. Circular Orbit

Optimum Thrust-to-(Mars) Weight Ratio * (non-cryogenic/cryogenic) - /2.0

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) - / 14,200

Total Starts 2

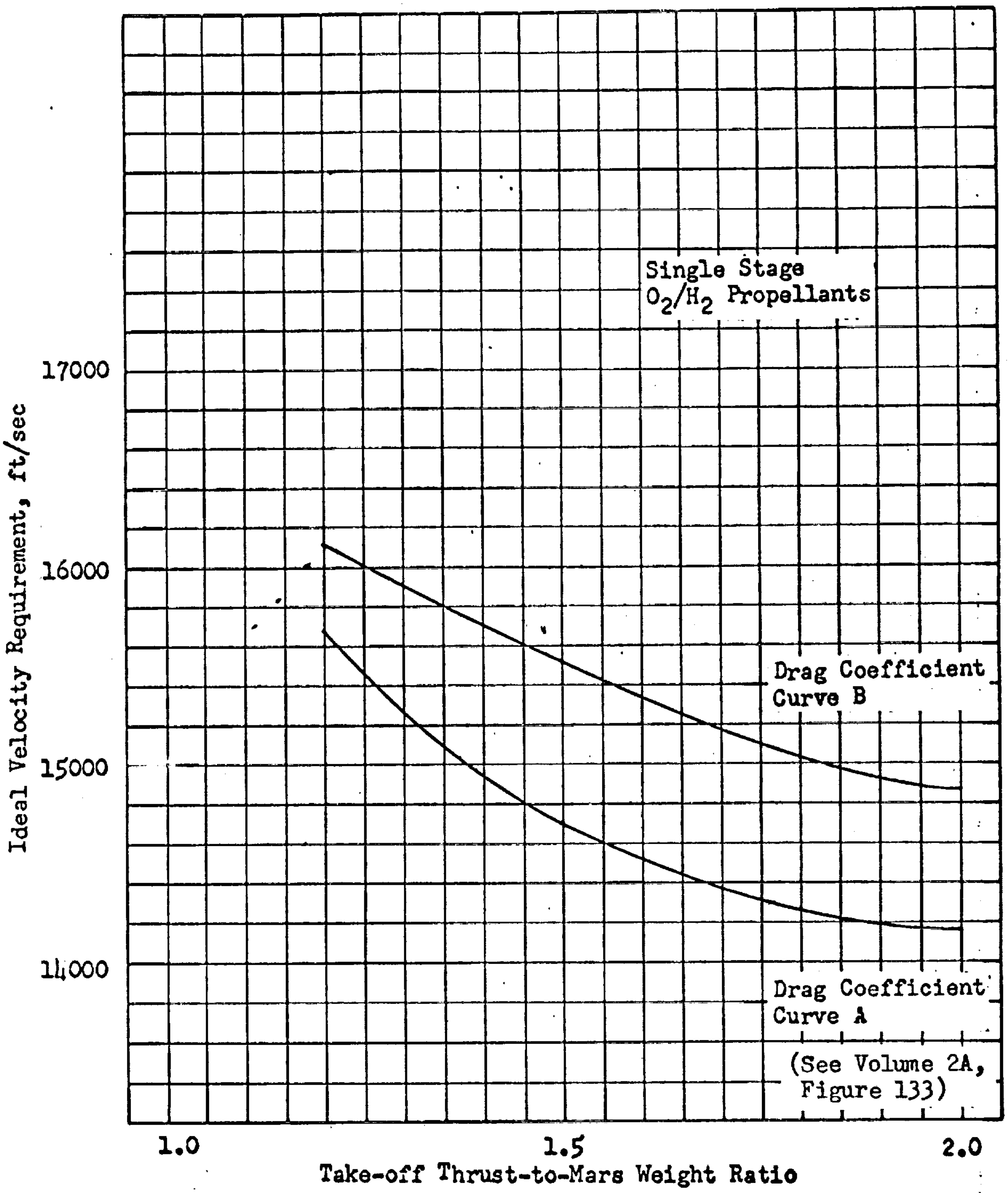
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	70	1150	1230
Expansion Area Ratio	120	200	150
Mixture Ratio	6.2	6.8	6.4
F_2/H_2 System			
Chamber Pressure, psia	105	1580	1780
Expansion Area Ratio	150	200	150
Mixture Ratio	17.0	17.4	17.2
N_2O_4/50-50 System			
Chamber Pressure, psia			
Expansion Area Ratio			
Mixture Ratio			

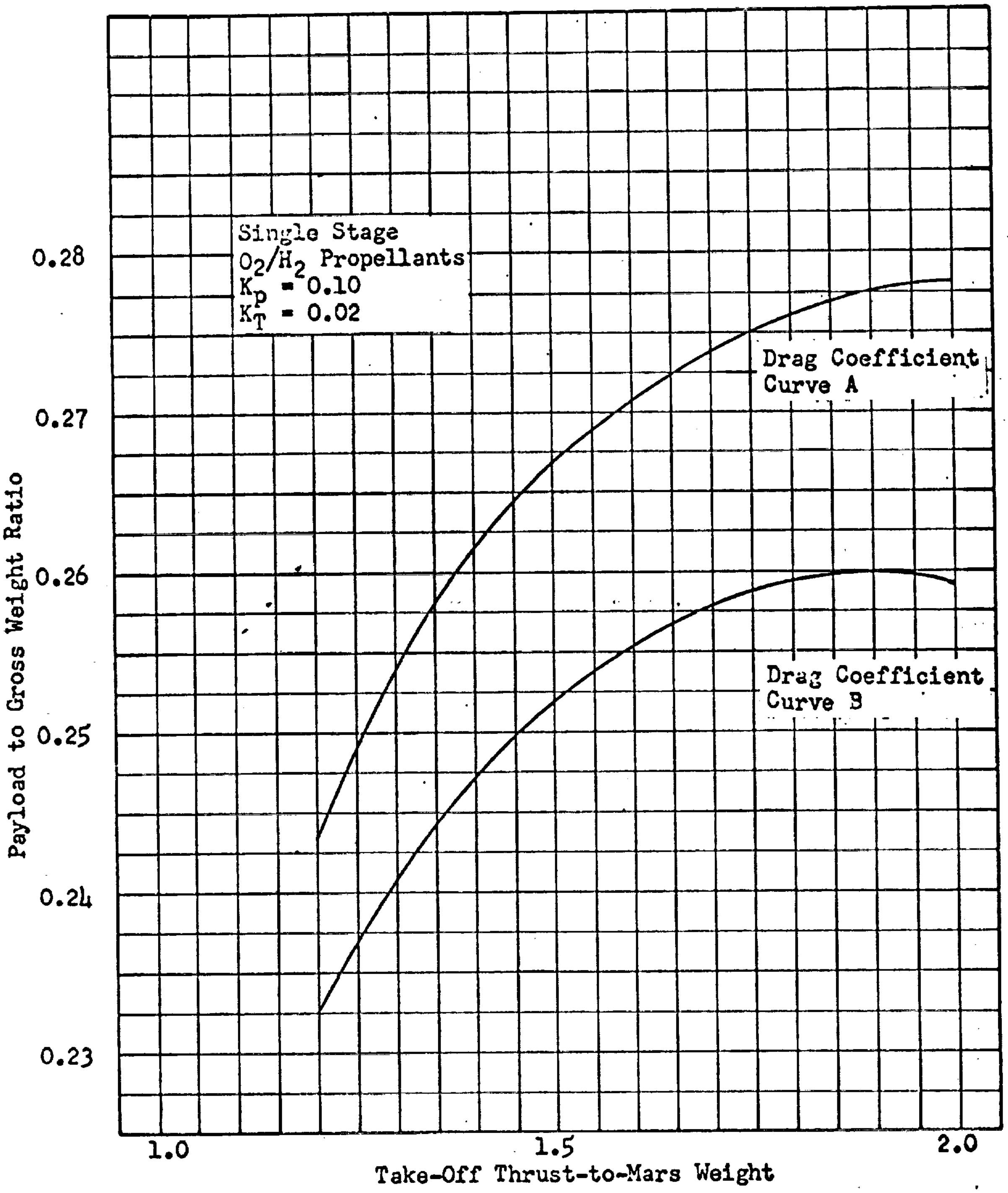
Related Figures

1. Ideal ΔV vs F/W
2. Payload to Gross Weight vs F/W

* Based on nonredundant engine system



Ideal Velocity Requirement Mars Take-Off to 300 n mi Circular Orbit.



Payload Mars Take-Off to 300 n m circular orbit

Planet: Venus

Maneuver: 300 N. Mi. Orbit Establishment from 12,000 ft/sec Hyperbolic Excess Velocity

Optimum Thrust-to-(Venus) Weight Ratio * (non-cryogenic/cryogenic) 0.40/0.40

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 11,590/11,950

Total Starts 1 - 3

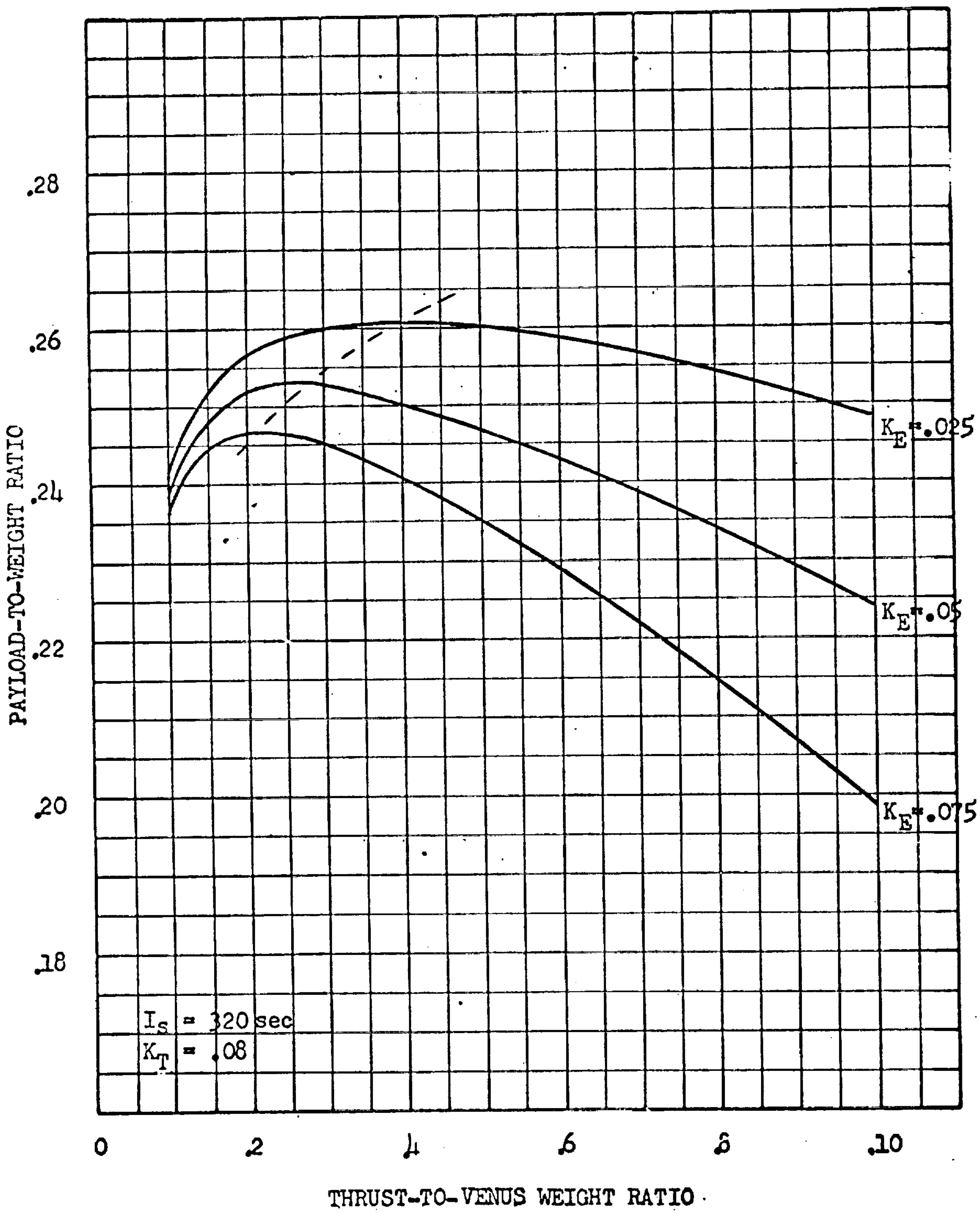
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	60	1340	1300
Expansion Area Ratio	190	400	190
Mixture Ratio	6.2	6.9	6.4
F_2/H_2 System			
Chamber Pressure, psia	90	1740	1880
Expansion Area Ratio	190	360	190
Mixture Ratio	16.9	17.3	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	140	1790	2000
Expansion Area Ratio	300	430	280
Mixture Ratio	2.2	2.2	2.2

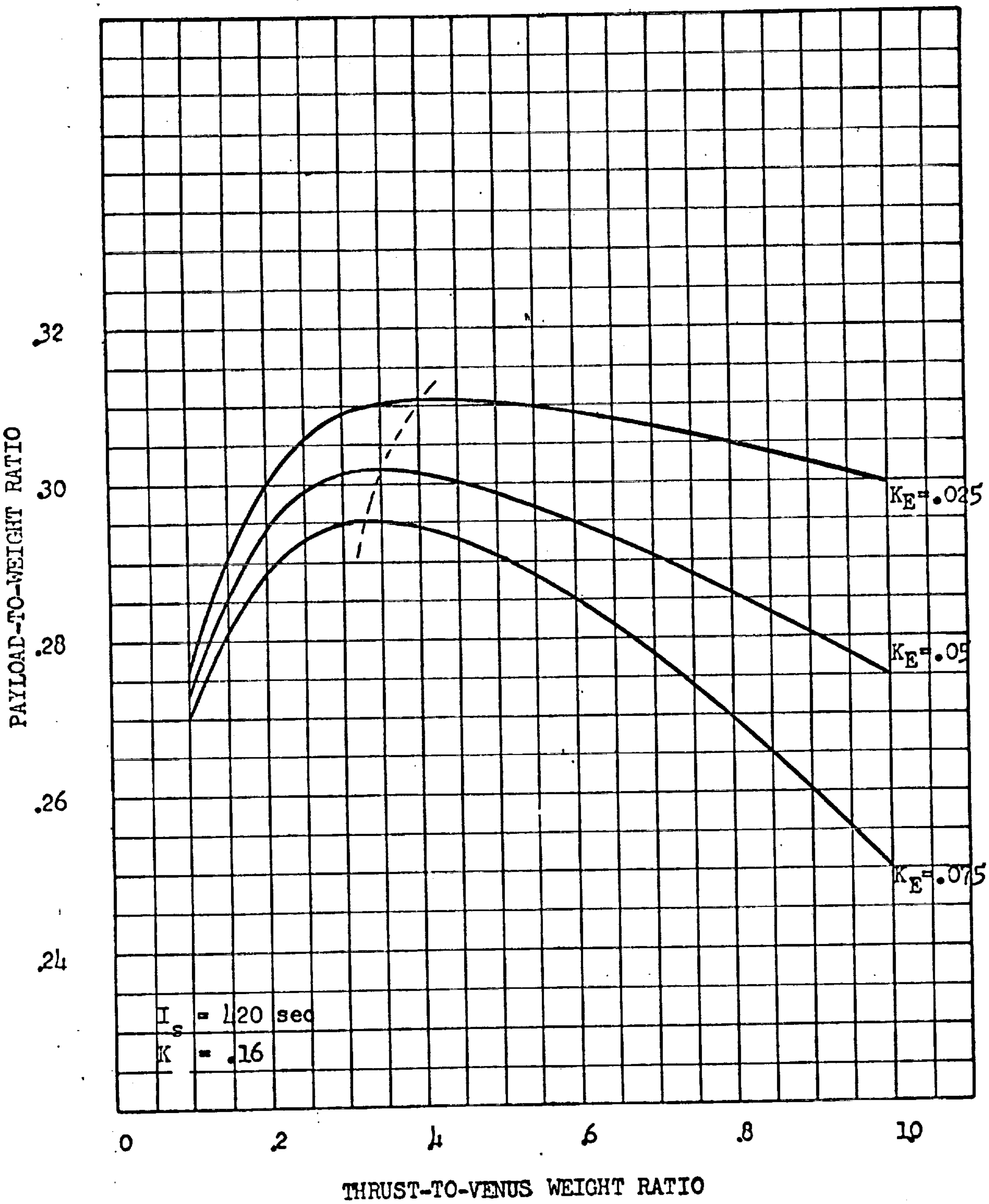
Related Figures

1. Payload to Gross Weight vs F/W; non-cryogenic
2. Payload to Gross Weight vs F/W; cryogenic
3. Payload to Gross Weight vs V_H and λ_p; cryogenic

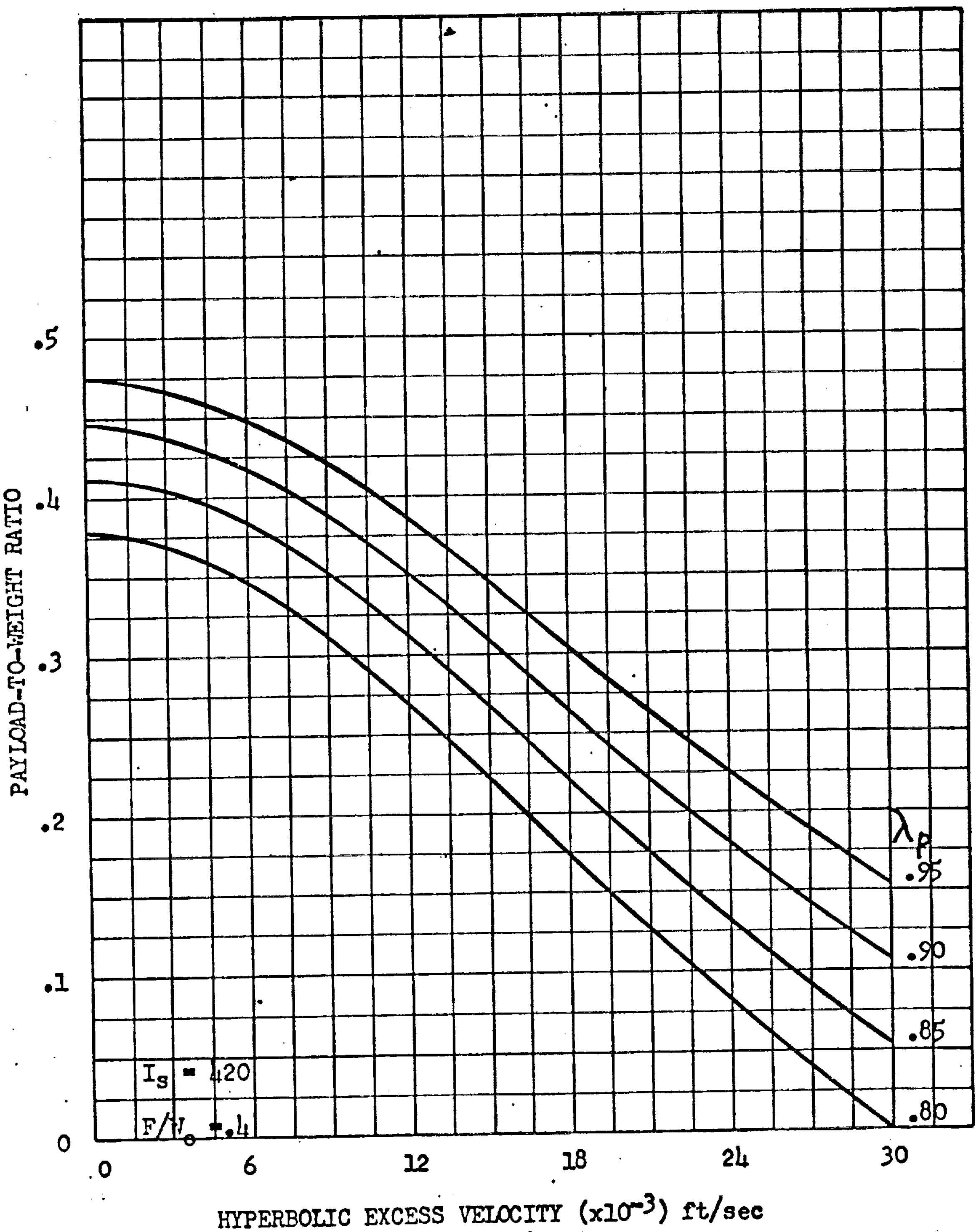
* Based on nonredundant engine system



VENUS - HYPERBOLIC EXCESS = 12,000 ft/sec



VENUS - HYPERBOLIC EXCESS = 12,000 ft/sec



VENUS PAYLOAD VS. HYPERBOLIC EXCESS

Planet: Moon

Maneuver: Direct Landing from 2.6 Days Transfer

Optimum Thrust-to-(Earth) Weight Ratio (non-cryogenic/cryogenic)* 0.40/0.45

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 10,000/9,900 (includes translation; 400 ft/sec, descent; 200 ft/sec, reserve; 300 ft/sec)

Total Starts 2

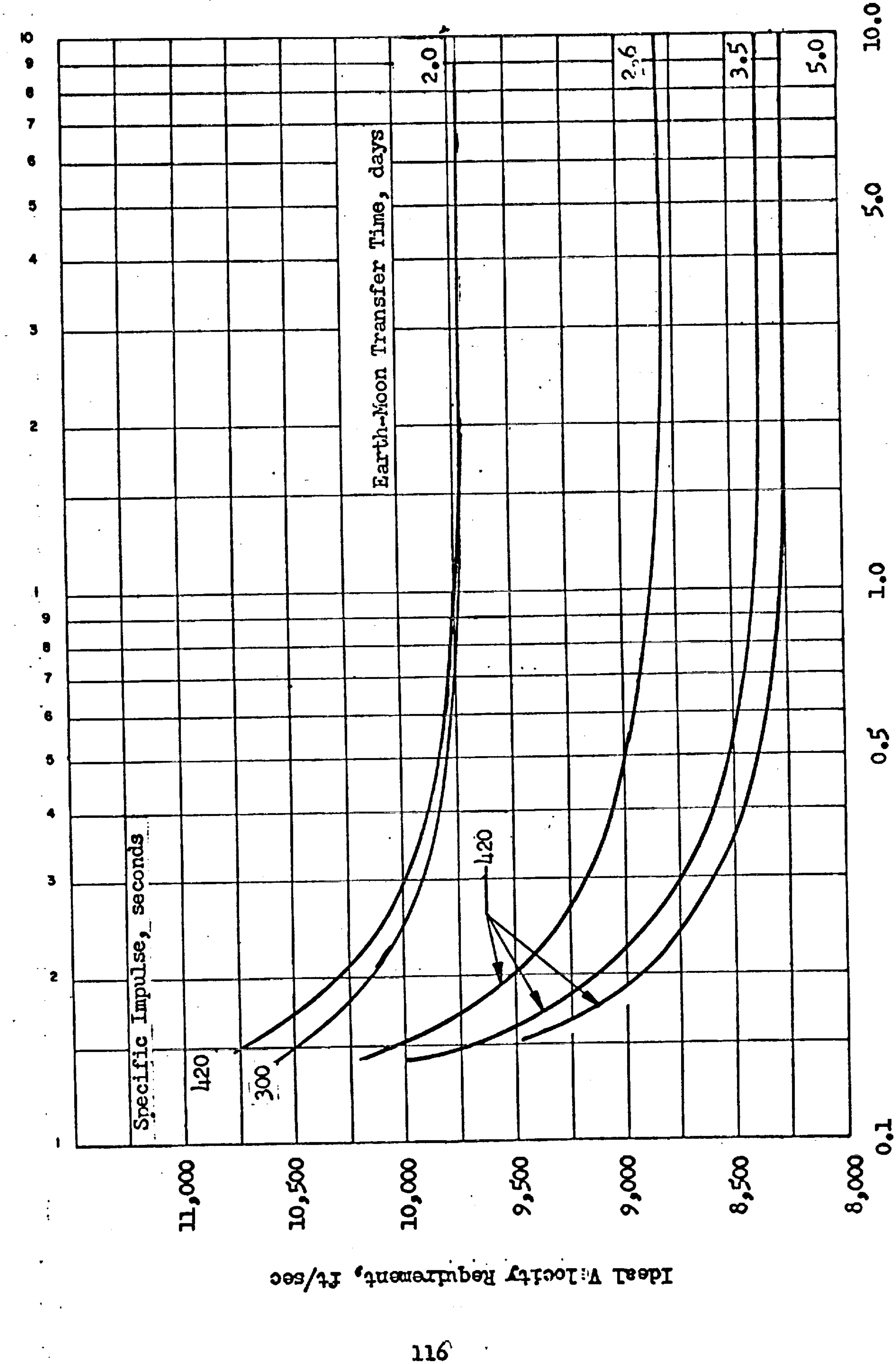
Throttling Ratio 10:1

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	70	1290	1300
Expansion Area Ratio	170	330	180
Mixture Ratio	6.2	6.8	6.4
F_2/H_2 System			
Chamber Pressure, psia	100	1700	1900
Expansion Area Ratio	180	300	180
Mixture Ratio	16.9	17.3	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	155	1790	2010
Expansion Area Ratio	290	400	270
Mixture Ratio	2.2	2.2	2.2

Related Figures

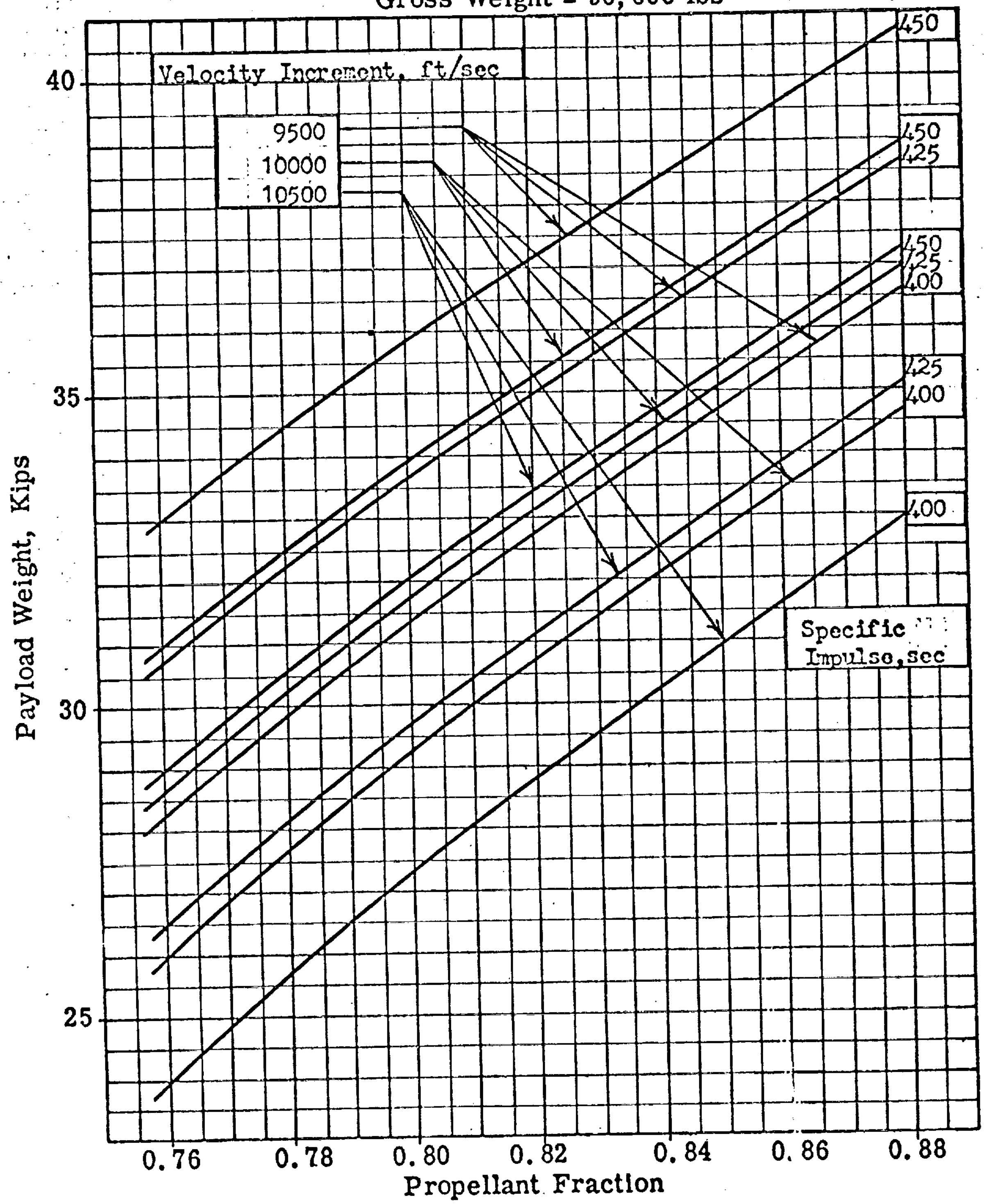
1. Ideal ΔV vs F/W
2. Payload vs λ_p, ΔV and I_s; cryogenic landing vehicles
3. Relative Gross Weight/Payload vs F/W

* Based on nonredundant engine system

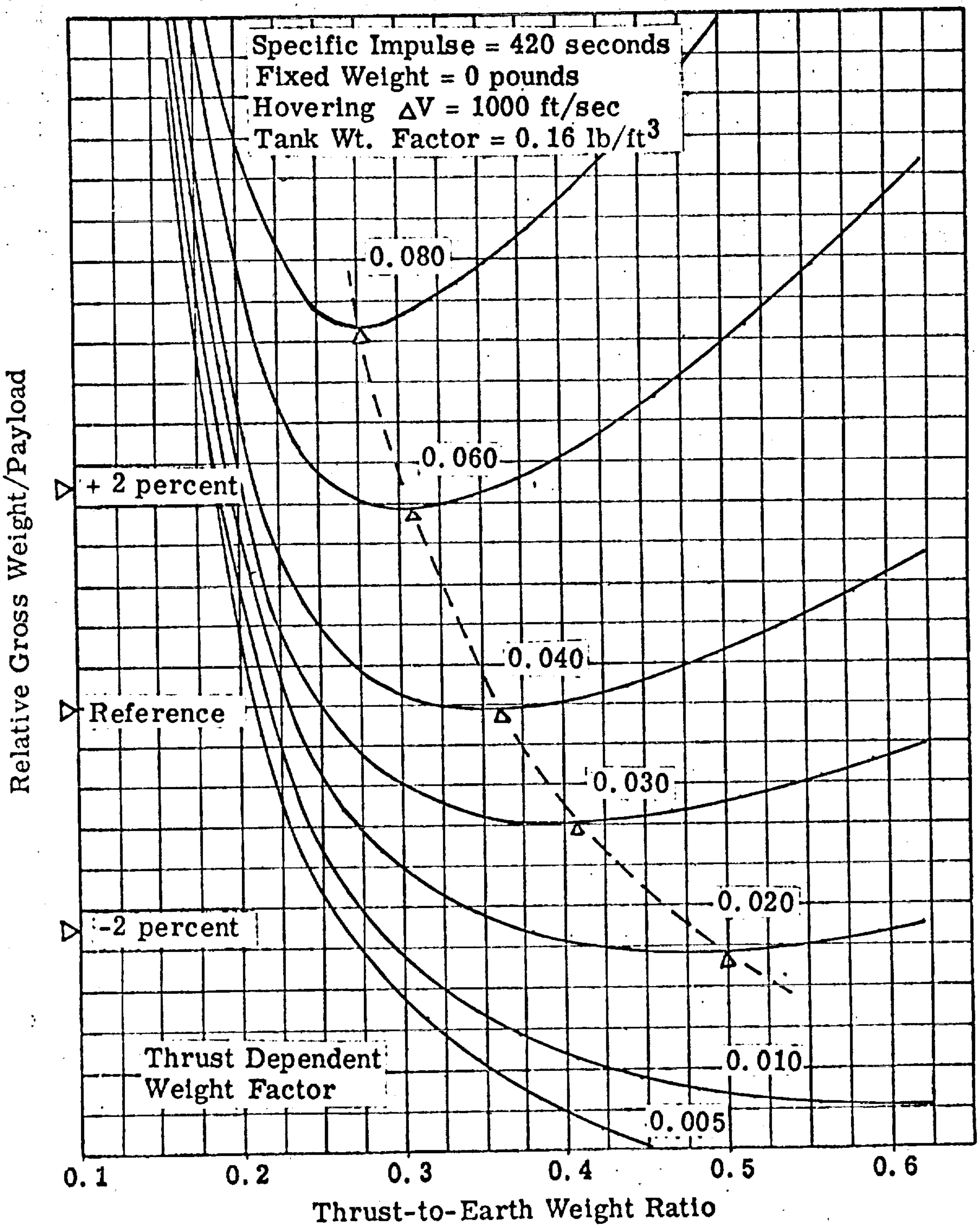


Ideal Velocity Requirement for Direct Lunar Landing From Earth-Moon Coast Trajectory; Trajectory Type C.

Gross Weight = 90,000 lbs



O_2/H_2 Lunar Landing Vehicles



Thrust Selection for Lunar Landing

Planet: Moon

Maneuver: Landing from 50 n mi. circular orbit

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) 0.55/0.55

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 6500/6500 (includes translation; 400 ft/sec, descent; 200 ft/sec, reserve; 200 ft/sec)

Total Starts 2

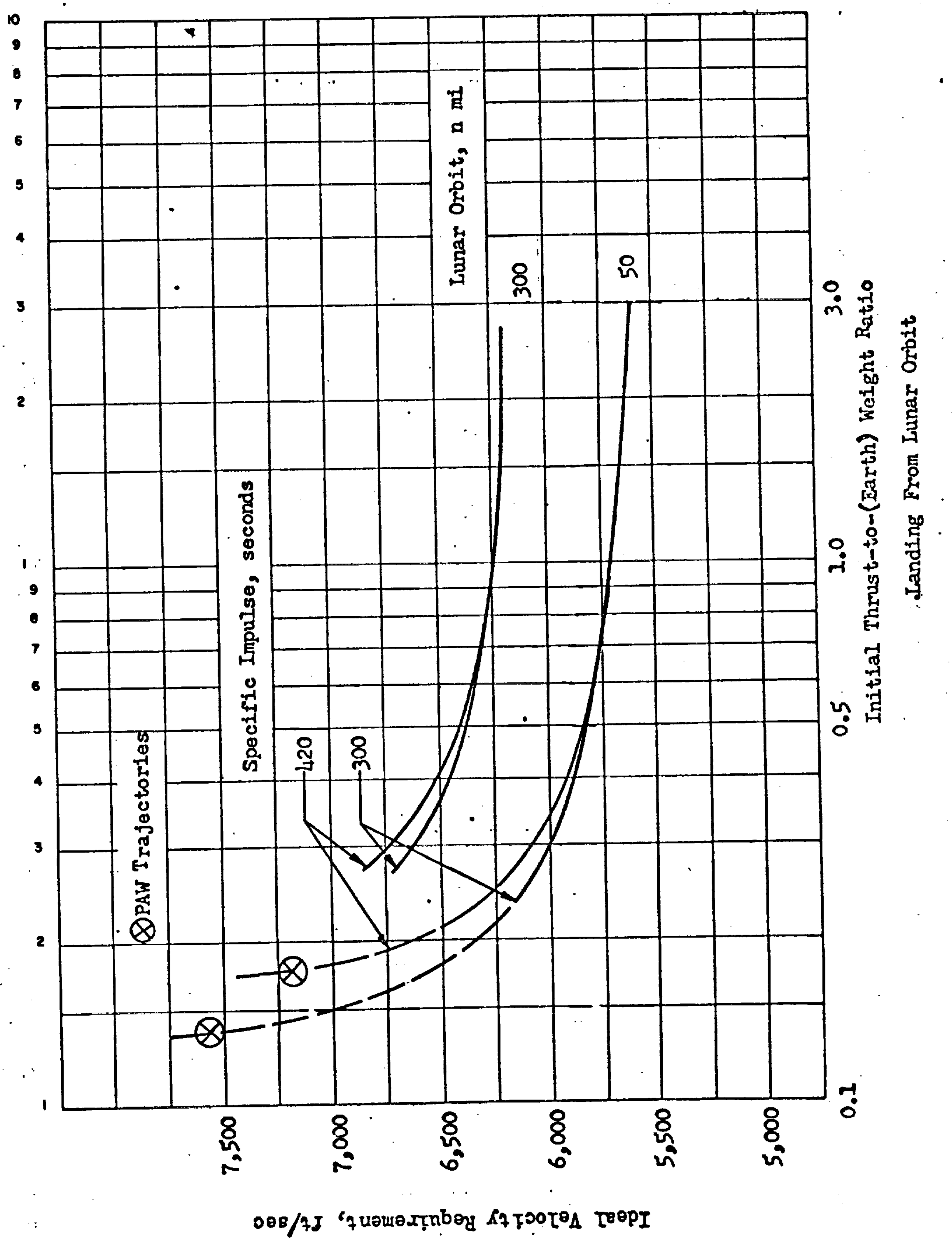
Throttling Ratio 10:1

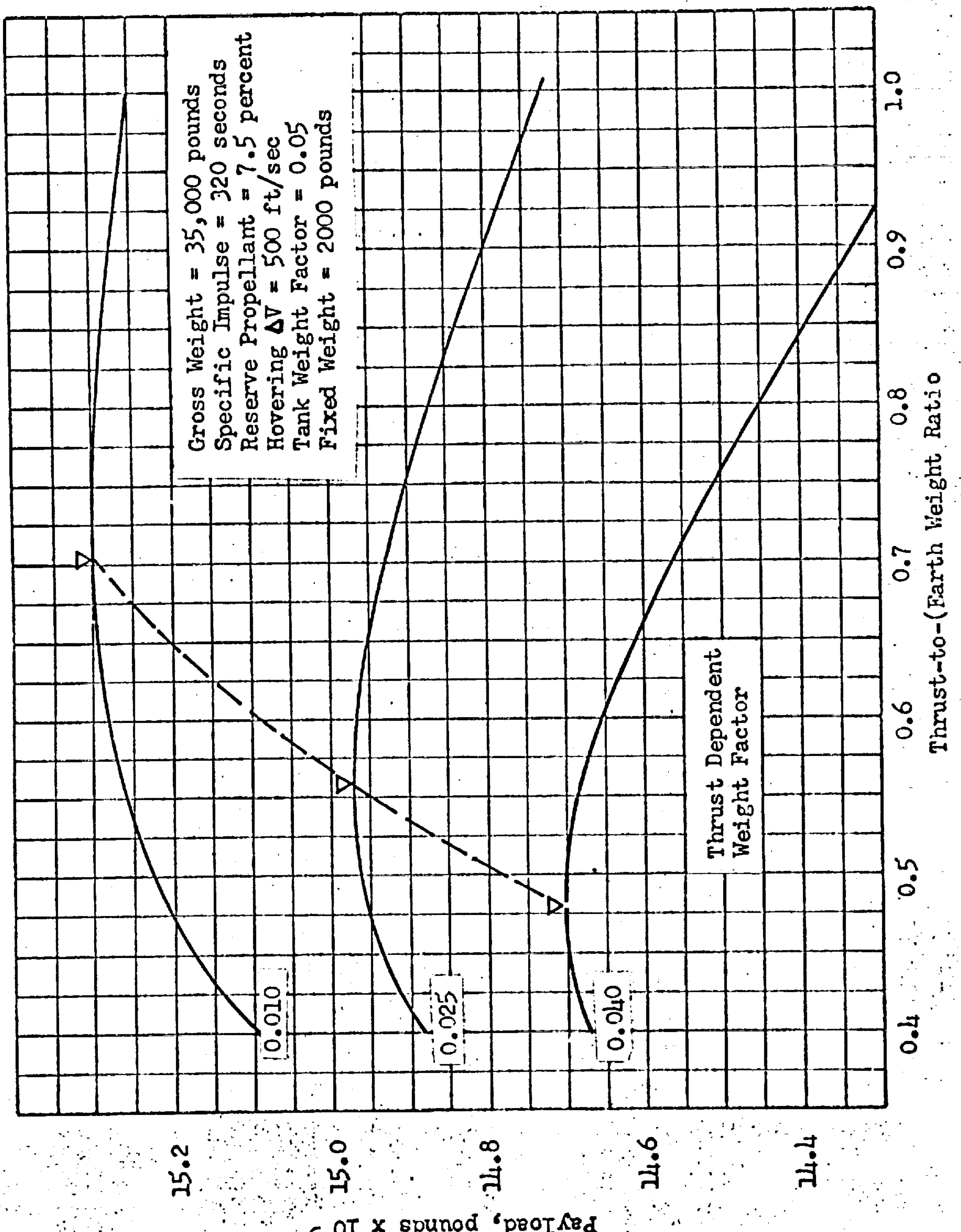
	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	80	1230	1280
Expansion Area Ratio	150	260	150
Mixture Ratio	6.1	6.7	6.3
F_2/H_2 System			
Chamber Pressure, psia	120	1650	1920
Expansion Area Ratio	170	230	160
Mixture Ratio	16.9	17.3	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	190	1750	2000
Expansion Area Ratio	260	320	240
Mixture Ratio	2.2	2.2	2.2

Related Figures

1. Ideal ΔV vs F/W and I_s
2. Payload vs F/W; non-cryogenic system

* Based on nonredundant engine system


Lunar Orbit, n mi
300
50
Specific Impulse, seconds
420
300
⊗ PAW Trajectories
Initial Thrust-to-(Earth) Weight Ratio
Landing From Lunar Orbit
0.1
0.5
1.0
3.0
7,500
7,000
6,500
6,000
5,500
5,000
Ideal Velocity Requirement, ft/sec



Thrust Selection for Landing from Lunar Orbit

Planet: Moon

Maneuver: Descent from and Ascent to 50 n mi Circular Orbit

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) 0.65/0.55

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 13,520/13,600 (includes 2000 ft/sec for translation, descent, plane-change and reserve allowances)

Total Starts 4

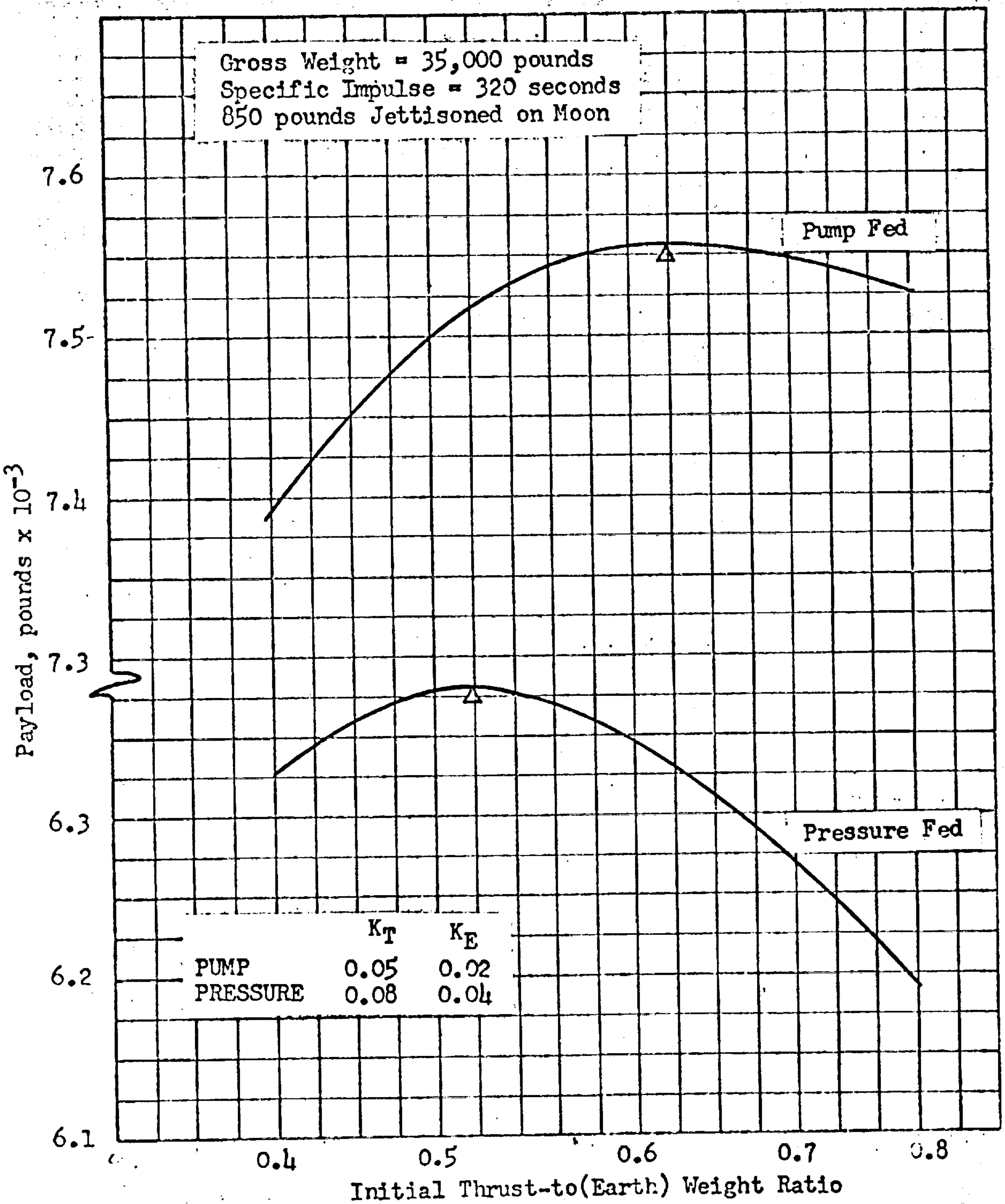
Throttling Ratio 10:1

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	60	1250	1260
Expansion Area Ratio	160	320	180
Mixture Ratio	62	6.8	6.4
F_2/H_2 System			
Chamber Pressure, psia	90	1670	1810
Expansion Area Ratio	170	300	180
Mixture Ratio	16.9	17.4	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	160	1710	1930
Expansion Area Ratio	230	300	230
Mixture Ratio	2.2	2.2	2.2

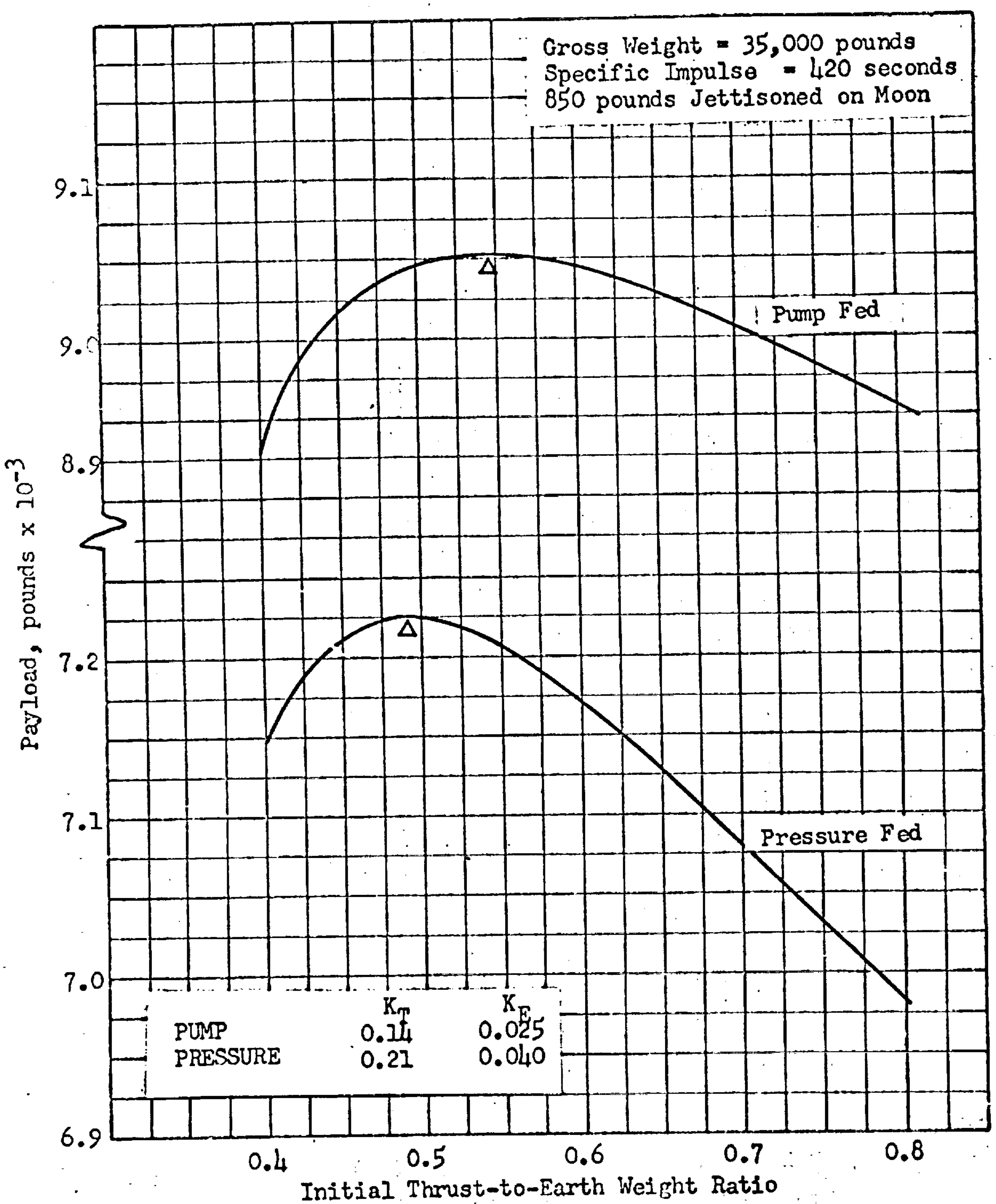
Related Figures

1. Payload vs Thrust-to-Weight Ratio; non-cryogenic
2. Payload vs Thrust-to-Weight Ratio; cryogenic

* Based on nonredundant engine system



Thrust Level Selection for Lunar Descent and Reorbit Vehicle



Thrust Level Selection for Lunar Descent and Reorbit Vehicle

Planet: Moon

Maneuver: Take-off to 50 n mi circular orbit

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) 0.75/0.65

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 7,000/7,040 (includes plane-change; 1000 ft/sec, reserve; 200 ft/sec)

Total Starts 2

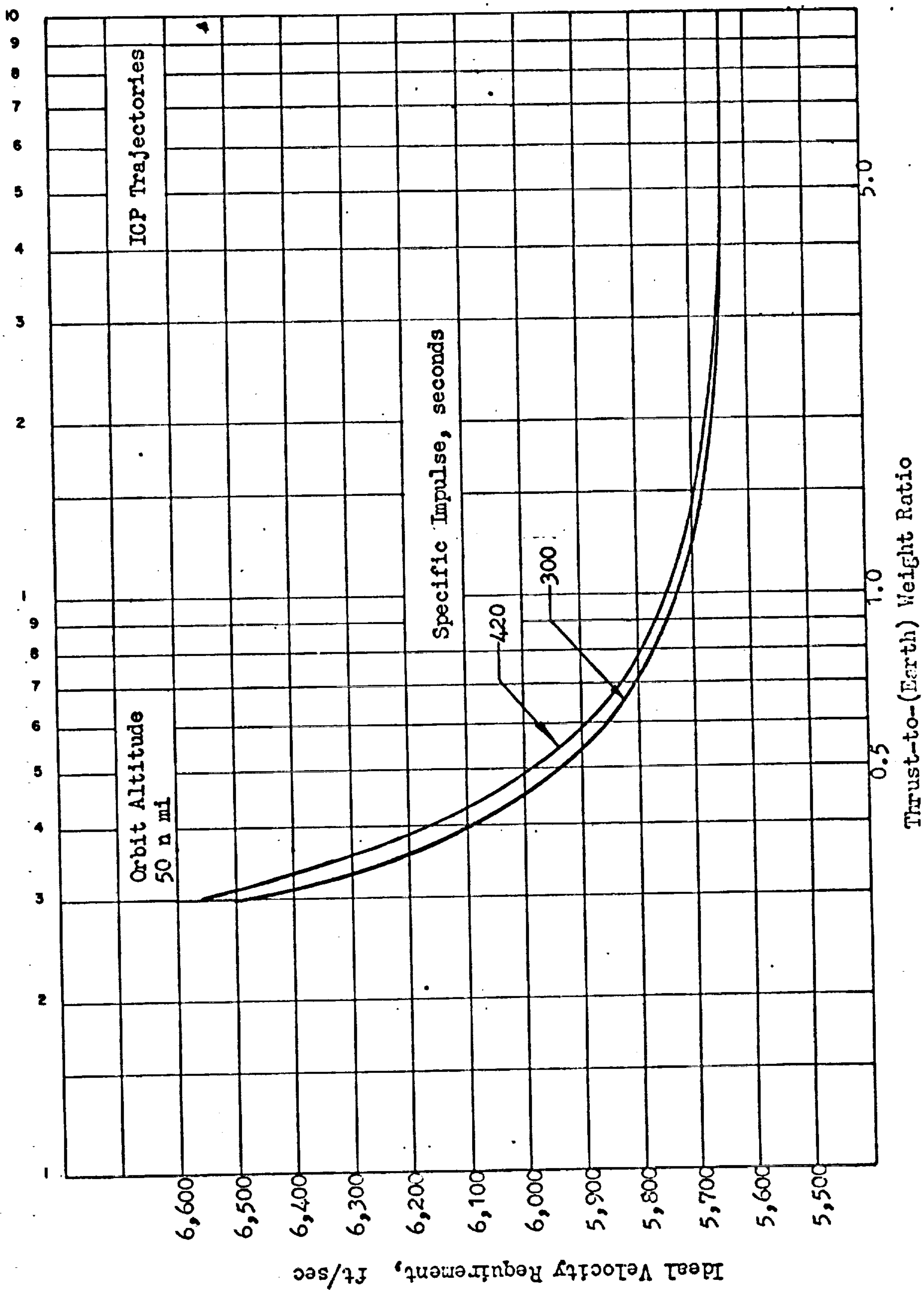
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	80	1180	1240
Expansion Area Ratio	140	220	130
Mixture Ratio	6.1	6.7	6.3
F_2/H_2 System			
Chamber Pressure, psia	125	1620	1870
Expansion Area Ratio	160	200	150
Mixture Ratio	17.0	17.3	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	210	1680	1950
Expansion Area Ratio	210	230	200
Mixture Ratio	2.2	2.2	2.2

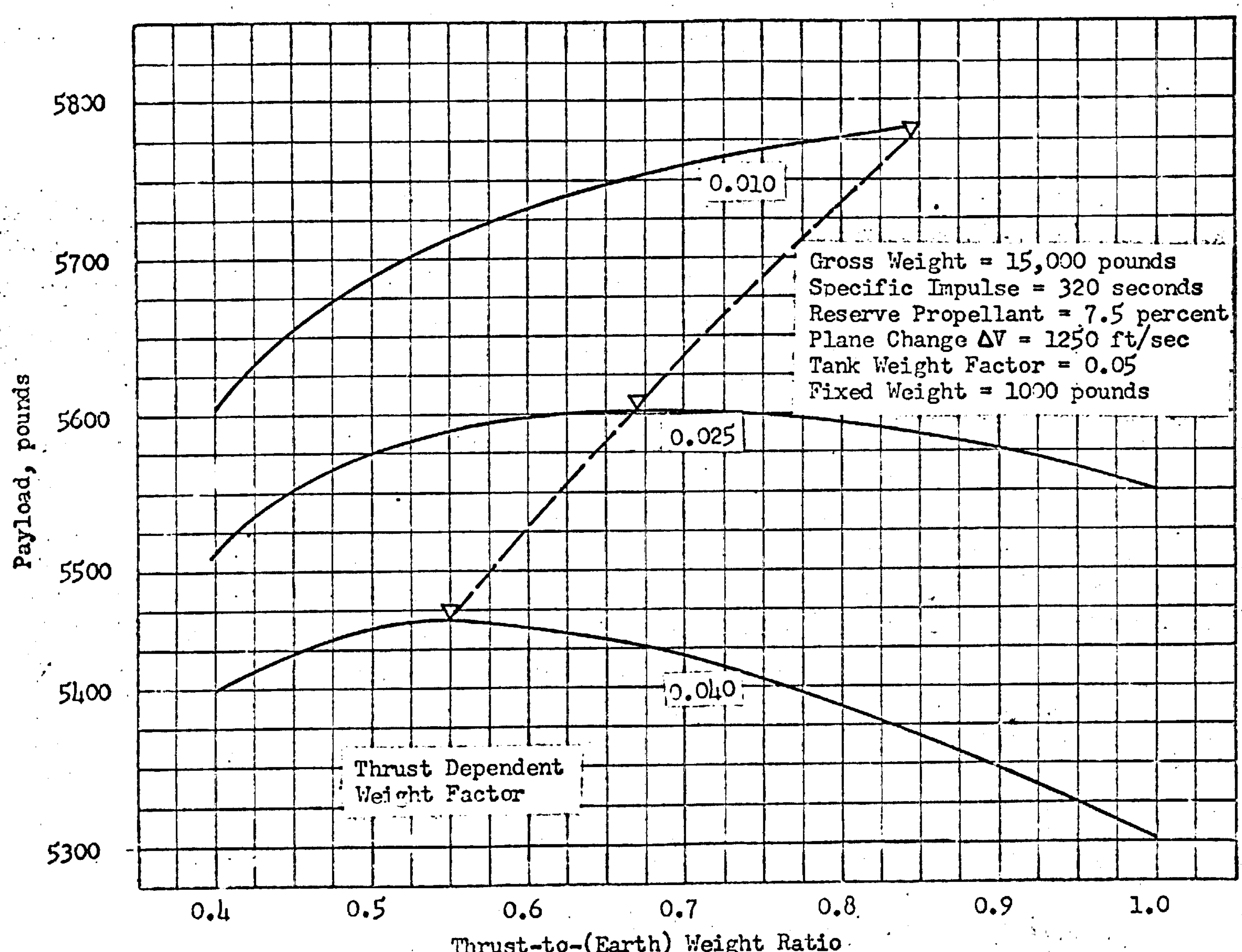
Related Figures

1. Ideal ΔV vs F/W
2. Payload vs F/W; non-cryogenic system
3. Direct Take-off Weight vs λ_p, ΔV and I_s; non-cryogenic
4. Direct Take-off Weight vs λ_p, ΔV and I_s; cryogenic

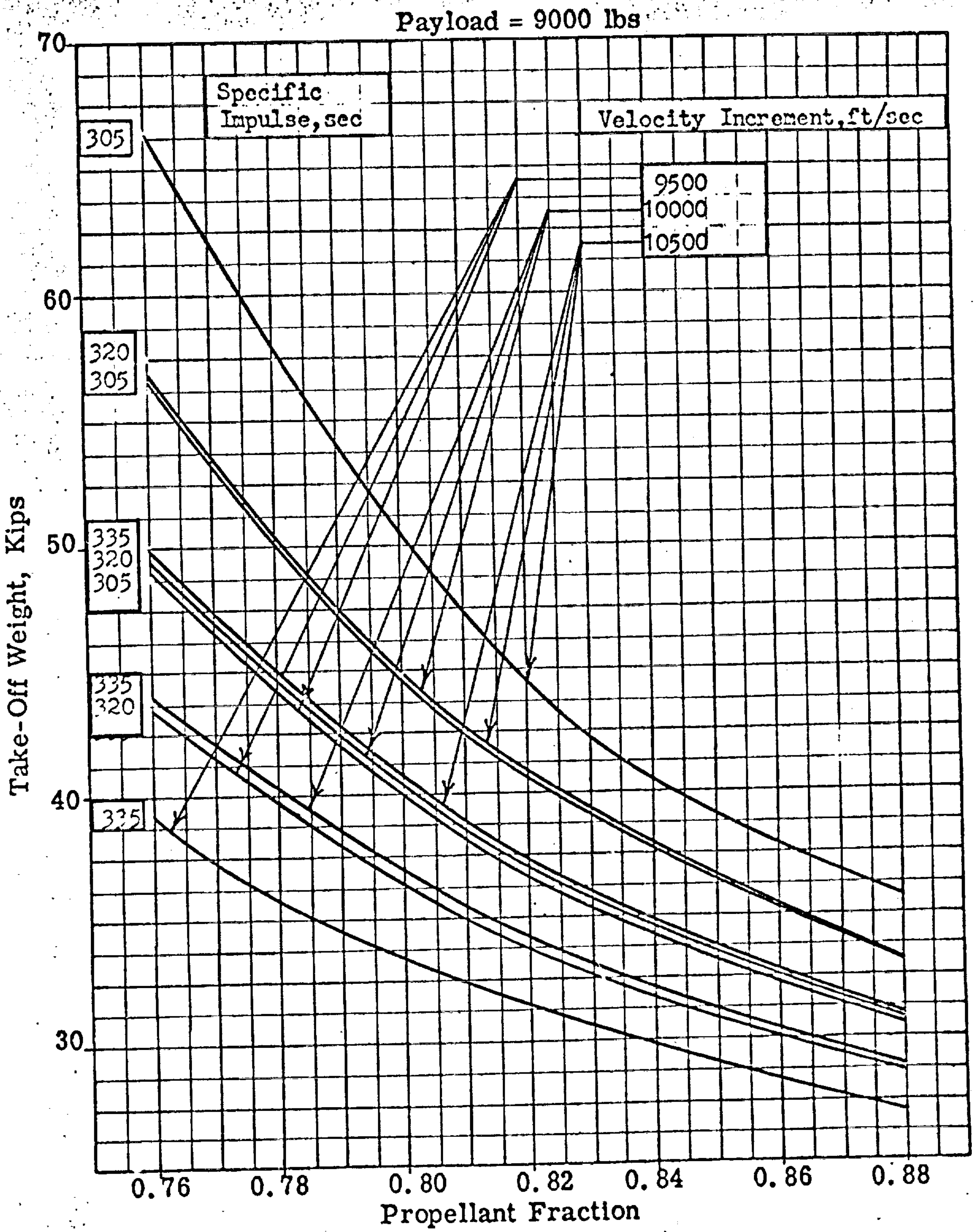
* Based on nonredundant engine system



50-n mi Lunar Orbit Establishment from Lunar Takeoff

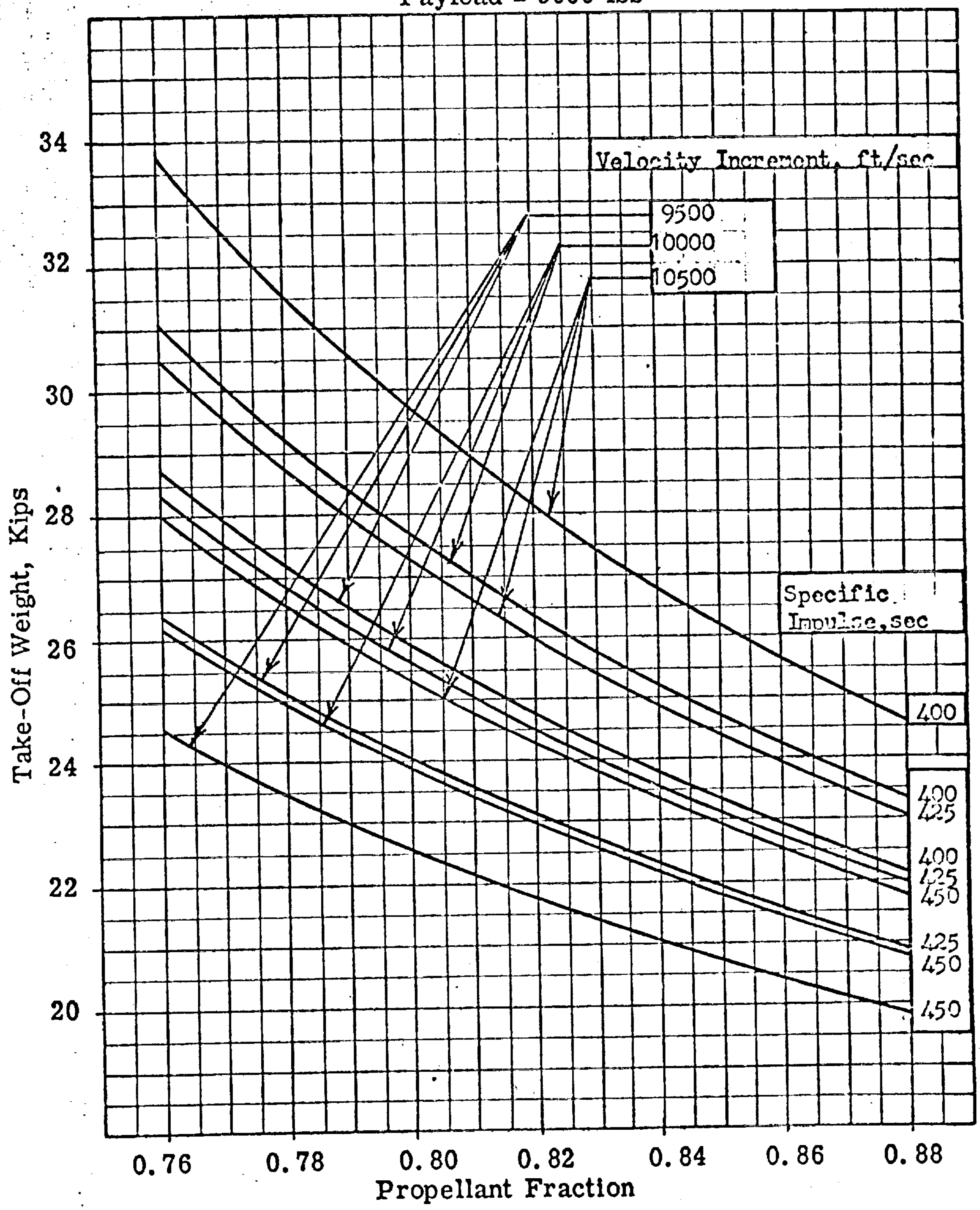


Thrust Selection for Lunar Takeoff-to-Orbit



Noncryogenic Propellant Lunar Take-Off Vehicles

Payload = 9000 lbs



O_2/H_2 Lunar Take-Off Vehicles

Planet: Mercury

Maneuver: 300 N Mi Circular Orbit Establishment from 30,000 ft/sec Hyperbolic Excess Velocity

Optimum Thrust-to-(Mercury) Weight Ratio * **(non-cryogenic/cryogenic)** /0.58

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) /25,100

Total Starts 1

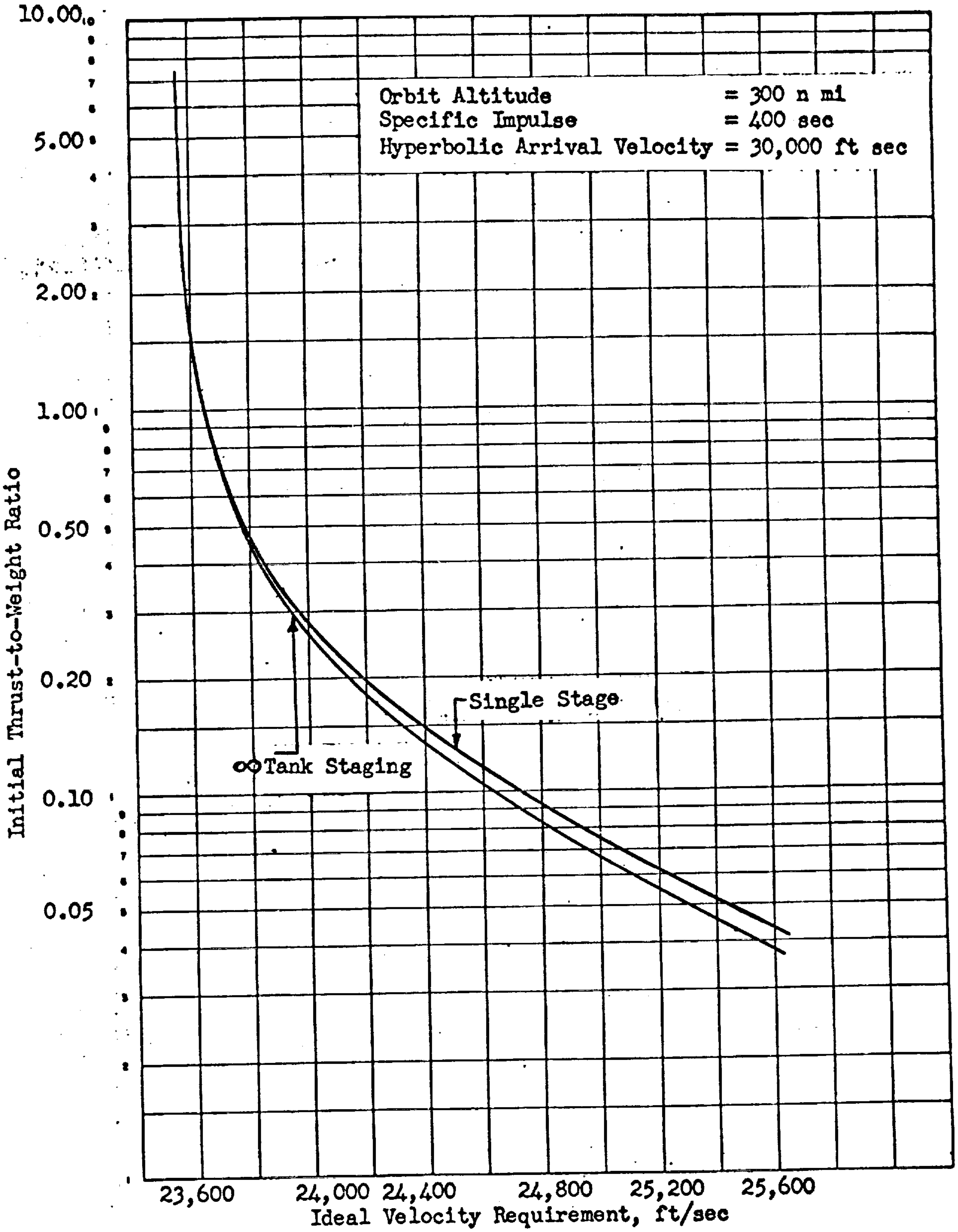
Throttling Ratio None Required

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	55	1400	1300
Expansion Area Ratio	200	490	210
Mixture Ratio	6.4	7.0	6.5
F_2/H_2 System			
Chamber Pressure, psia	60	1760	1920
Expansion Area Ratio	210	410	210
Mixture Ratio	17.1	17.5	17.2
N_2O_4/50-50 System			
Chamber Pressure, psia			
Expansion Area Ratio			
Mixture Ratio			

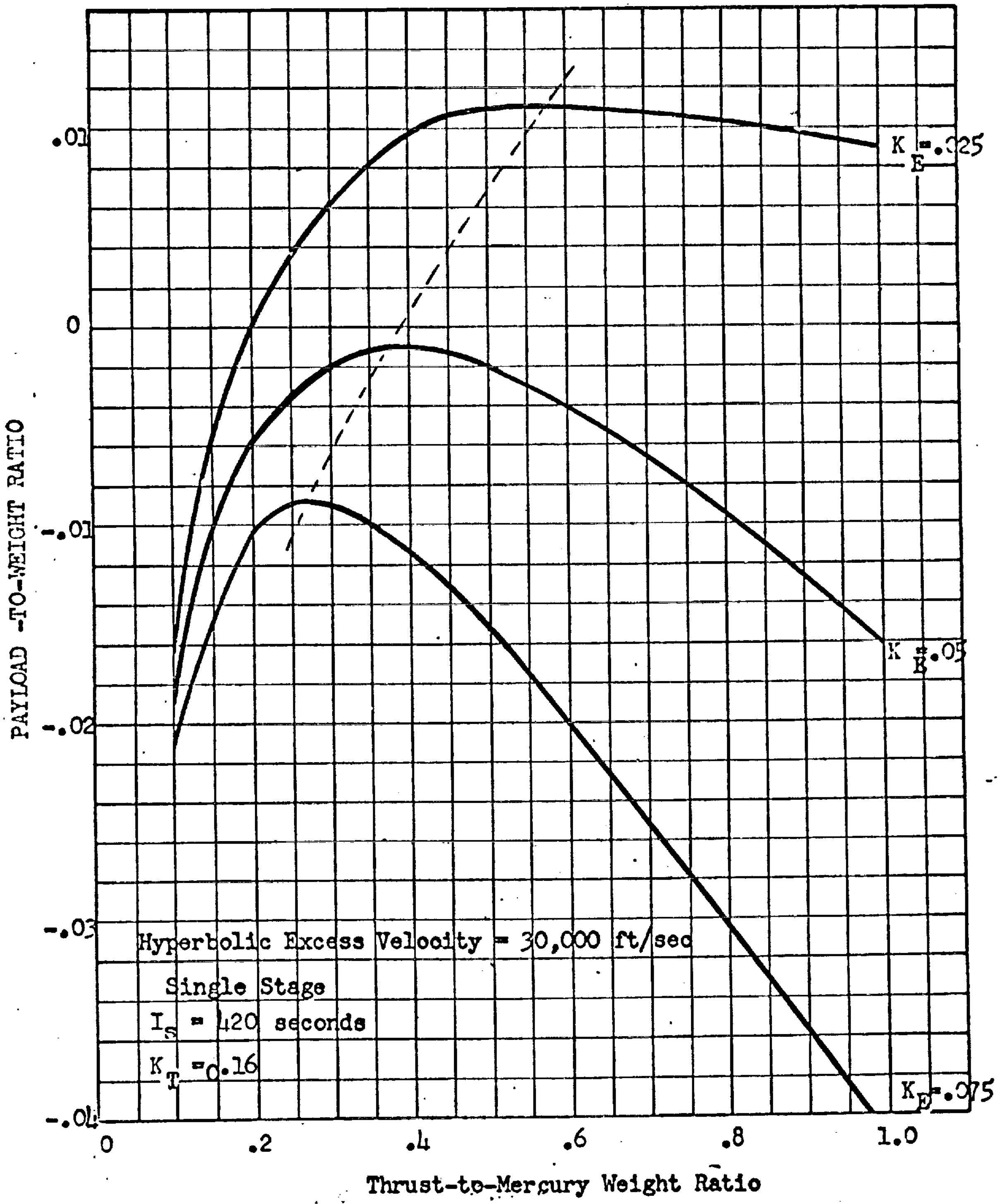
Related Figures

1. Ideal Velocity Requirements vs Thrust-to-Weight Ratio
2. Payload to Gross Weight vs. F/W (Single Stage)

* Based on nonredundant engine system



Mercury Orbit Establishment
Ideal Velocity Requirement



Orbit Establishment Maneuver Thrust-to-Weight Optimization for Mercury

Planet: Mercury

Maneuver: Landing from 300 N Mi Circular Orbit

Optimum Thrust-to-(Earth) Weight Ratio * (non-cryogenic/cryogenic) 0.80/0.85

Ideal Velocity Requirements, ft/sec (noncryogenic/cryogenic) 12,500/12,400 (includes translation; 600 ft/sec, descent 300 ft/sec, reserve; 400 ft/sec)

Total Starts 2

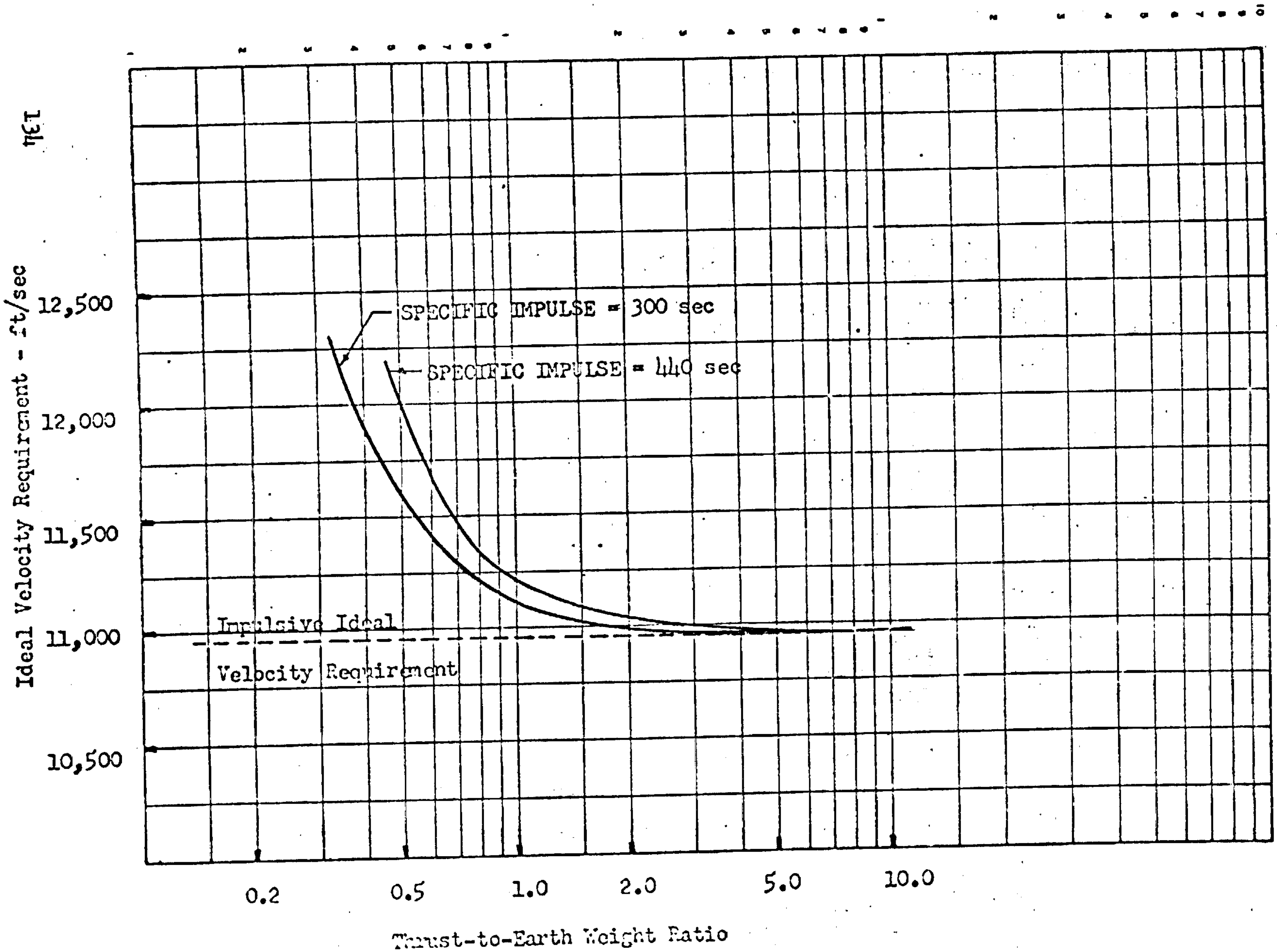
Throttling Ratio 9:1

	Optimum Propulsion Parameters		
	5,000-pound Thrust System	50,000-pound Thrust System	500,000-pound Thrust System
O_2/H_2 System			
Chamber Pressure, psia	75	1110	1210
Expansion Area Ratio	110	170	130
Mixture Ratio	6.2	6.7	6.4
F_2/H_2 System			
Chamber Pressure, psia	110	1550	1770
Expansion Area Ratio	140	180	140
Mixture Ratio	17.0	17.4	17.1
N_2O_4/50-50 System			
Chamber Pressure, psia	180	1670	1920
Expansion Area Ratio	200	210	200
Mixture Ratio	2.2	2.2	2.2

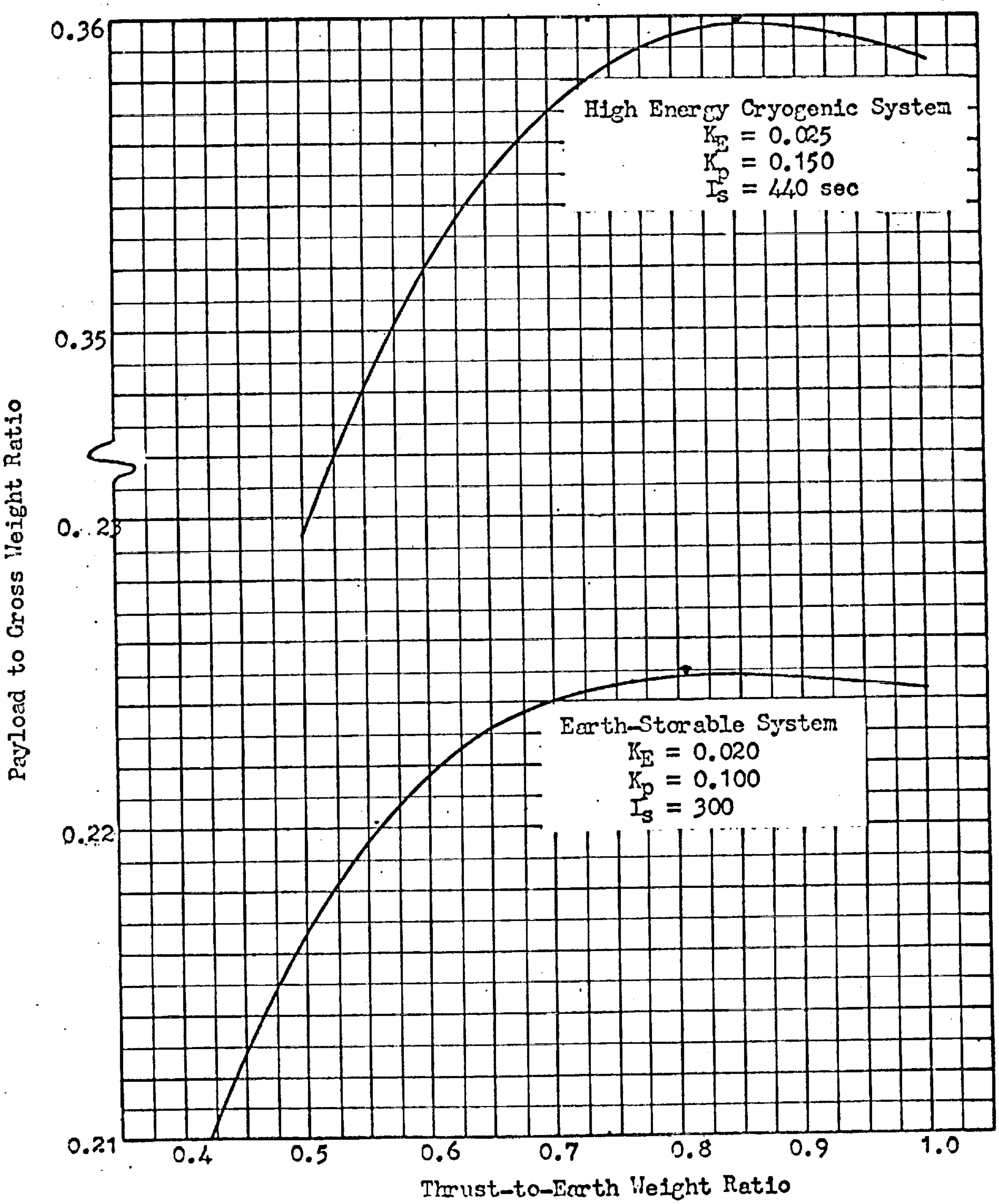
Related Figures

1. Ideal ΔV vs F/W and I_s
2. Payload to Gross Weight vs F/W

* Based on nonredundant engine system



Ideal Velocity Requirements for a Planet Mercury Landing From a 300 N. Mi. Circular Orbit



Mercury Landing from 300 N Mi Orbit

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4. R 2151P, Design Studies, 200,000 Pound Thrust Oxygen/Hydrogen Propulsion System, Rocketdyne, A Division of North American Aviation, Inc., March 1960.

5. R 3553, Engine-Out Capability, Rocketdyne, A Division of North American Aviation, Inc., April 1962.